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RESEARCH MEMORANDUM

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for the

Air Research and Development Command, U. S. Air Force

OVER-ALL PERFORMANCE OF J65-B3 TURBOJET ENGINE FOR REYNOLDS NUMBER INDICES FROM 0.8 TO 0.2

By David B. Fenn and William L. Jones

Lewis Flight Propulsion Laboratory
Cleveland, Ohio

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OVER-ALL PERFORMANCE OF J65-B3 TURBOJET ENGINE FOR REYNOLDS NUMBER INDICES FROM 0.8 TO 0.2

By D. B. Fenn and William L. Jones

SUMMARY

The steady-state over-all performance characteristics of the J65-B3 turbojet engine were determined in an altitude test chamber for four exhaust-nozzle areas at Reynolds number indices of 0.8, 0.4, and 0.2. This range of Reynolds number indices corresponds to a range of altitudes from about sea level to 51,500 feet at a flight Mach number of 0.8.

Generalized data are presented to allow calculation of engine performance at any flight condition corresponding to a Reynolds number index within the range investigated. Engine performance calculated from these generalized data is presented for seven altitudes over a range of flight speeds from zero to about 1100 knots.

The use of an exhaust nozzle sized to give rated performance at sea level would permit operation near the point of minimum specific fuel consumption for a wide range of flight conditions and engine speeds.

INTRODUCTION

The performance characteristics of the J65-B3 turbojet engine were determined in an altitude test chamber at the NACA Lewis laboratory at the request of the Air Research and Development Command, U. S. Air Force. Preliminary altitude performance data with the rated exhaust-nozzle area, together with the operational limits and windmilling and starting characteristics of the engine, are contained in reference 1. The component performance of the engine is presented in reference 2, and the rotating-stall and blade-vibration characteristics of the compressor are reported in reference 3.

The present report summarizes the over-all performance of the engine by presenting performance for a range of exhaust-nozzle areas, altitudes, and flight Mach numbers. Data are presented in terms of conventional generalized parameters for a range of corrected engine speeds from 6700 to 9200 rpm, Reynolds number indices from 0.8 to 0.2, and exhaust-nozzle areas from 1.90 to 2.51 square feet. These data permit calculation of performance at any operating condition where sonic (choked) flow exists in the exhaust nozzle. Over-all engine performance maps are presented for a flight Mach number of 0.7 at altitudes of 15,000, 35,000, and 50,000 feet. In addition, the altitude performance of the engine, calculated from generalized data for the rated exhaust nozzle (1.97 sq ft), is presented for a range of altitudes from sea level to 55,000 feet, flight speeds from zero to 1100 knots, and engine speeds from 7000 to 8300 rpm. The data are presented in both graphical and tabular form.

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APPARATUS

Engine

The J65-B3 turbojet engine (fig. 1) has a 13-stage axial-flow compressor, an annular prevaporizing-type combustion chamber, and a two-stage turbine. At military rated conditions, the engine speed is 8300 rpm and the turbine-discharge temperature is 1166° F. At sea-level static conditions with no compressor-inlet screen, the engine has a guaranteed thrust of 7220 pounds and specific fuel consumption of 0.92. The sea-level, static, rated air flow of the engine is approximately 118 pounds per second. The engine is $87\frac{5}{8}$ inches long from the compressor-inlet flange to the turbine exit and has a maximum diameter of $37\frac{3}{4}$ inches. The engine dry weight is 2785 pounds. The fuel used throughout this investigation was MIL-F-5624A, grade JP-4.

Installation

The engine was installed in an altitude test chamber as shown in figure 1. A bulkhead with a labyrinth seal around the front of the engine (fig. 2) was used to allow independent control of inlet and exhaust pressures. The laboratory air systems supplied combustion air to the engine and removed the exhaust gases. The engine was mounted on a thrust platform equipped with a null-type pneumatic balance.

Instrumentation

The location and amount of instrumentation used during this investigation are shown in figure 2. Total-pressure and -temperature probes were located at the centers of equal annular areas at various stations in the engine. Engine fuel flow was measured by calibrated rotometers. All pressures were measured with manometers and recorded photographically. Self-balancing potentiometers were used to record all temperatures.

PROCEDURE

With each of the four exhaust nozzles (1.90, 1.97, 2.07, and 2.51 sq ft) investigated, the engine was operated at inlet conditions corresponding to Reynolds number indices of 0.8, 0.4, and 0.2. At each Reynolds number index, the ram pressure ratio (P_2/p_0) was set close to the facility limit with the engine operating either at rated speed or limiting turbine-discharge temperature. The ram pressure ratio was then held constant while engine speed was reduced. This procedure was used in order to maintain sonic flow in the exhaust nozzle over as wide an engine-speed range as possible. When the exhaust nozzle is fully choked, the pressures and temperatures within the engine are independent of the ambient pressure. The symbols and methods of calculation used in this report are presented in appendixes A and B, respectively.

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RESULTS AND DISCUSSION

Performance data are presented in terms of generalized parameters to show the effects of Reynolds number and to allow calculation of performance at specific flight conditions. To summarize the performance of this engine, performance maps were calculated from generalized data for the rated exhaust nozzle over a wide range of flight conditions.

The exhaust-nozzle flow coefficients reported for this engine in reference 2 are essentially constant for nozzle pressure ratios in excess of 2.0. The corrected engine speed at which the exhaust-nozzle pressure ratio is equal to 2.0 is presented as a function of flight Mach number in figure 3 for the four exhaust nozzles investigated. Operation above the line for each nozzle area (choked) indicates the region where the exhaust-nozzle pressure ratio is 2.0 or higher.

The operational limits of the engine with the rated exhaust nozzle at a flight Mach number of 0.8 are reproduced from reference 1 in figure 4. The operational limits include the engine speed for limiting turbine-discharge temperature, "idle" throttle position, lean combustion blow-out, and windmilling as functions of altitude. It can be seen from figure 4 that limiting turbine-discharge temperature occurred below rated engine speed for altitudes above 20,000 feet. The throttle position specified by the manufacturer as "idle" limited operation to altitudes below 67,000 feet. However, reductions in fuel flow below "idle" permitted steady-state operation up to a facility limit encountered at

75,000 feet. The shaded area superimposed on figure 4 illustrates the range of conditions over which performance can be calculated from the generalized data presented herein for the rated exhaust-nozzle area. The extent of this region is limited to altitudes below that corresponding to an inlet Reynolds number index of 0.2 and to engine speeds above that required to reach an exhaust-nozzle pressure ratio of 2.0. This shaded area covers the major portion of the practical operating conditions of the engine within the range of Reynolds number indices investigated. The numerical examples contained in appendix C illustrate calculation procedures that may be used to obtain performance within the 3631
region where the exhaust-nozzle flow coefficient is constant. If performance data are required in the region where the exhaust-nozzle flow coefficient is not constant, the method reported in reference 4 may be used.

Generalized Performance

Air flow. - Corrected engine inlet air flow is presented as a function of corrected engine speed in figure 5 for a range of Reynolds number indices from 0.8 to 0.2 and exhaust-nozzle areas from 1.90 to 2.51 square feet. Within the range of variables included in this investigation, no consistant variation of corrected air flow with either Reynolds number index or exhaust-nozzle area could be detected. At sea-level static and military rated conditions the air flow of the engine used for this investigation was about 4 percent higher than the manufacturer's specifications.

Pumping characteristics. - The variation of engine total-pressure ratio with corrected engine speed at a Reynolds number index of 0.4 is shown in figure 6 for the four exhaust nozzles investigated. Lines for exhaust-nozzle areas other than those included in this investigation and lines of constant engine temperature ratio have been cross-plotted onto this figure to facilitate calculation of engine performance at specific flight conditions. An inlet Reynolds number index of 0.4 was selected for the presentation of these engine pumping characteristics because the widest range of corrected engine speeds was obtained at this condition.

The effects of Reynolds number on engine pumping characteristics are shown in figure 7. In this figure engine total-pressure and -temperature ratios are divided by their respective values for the same corrected engine speed and exhaust-nozzle area at a Reynolds number index of 0.4. Although it is not generally possible to draw single curves to show Reynolds number effects on pumping characteristics, in this instance the variations with both corrected engine speed and exhaust-nozzle area were less than 1 percent from the mean. As Reynolds number index is decreased from 0.8 to 0.2 at a given corrected engine speed and exhaust-nozzle area, the engine total-pressure and -temperature ratios increased approximately 4 and 8 percent, respectively.

Thrust. - As shown in reference 5, the jet thrust obtained from an engine with a choked exhaust nozzle can be correlated by the following relation:

$$\frac{F_j}{A_n} = C_F\left[(\gamma + 1)\left(\frac{2}{\gamma + 1}\right)^{\frac{\gamma}{\gamma-1}} p_9 - p_0\right] \tag{1}$$

For nonafterburning operation, the average value of $(\gamma + 1)\left(\frac{2}{\gamma + 1}\right)^{\frac{\gamma}{\gamma-1}}$ was found to be 1.26.

The correlation of the jet thrust per unit of exhaust-nozzle area is presented in figure 8 for a range of Reynolds number indices from 0.8 to 0.2 and exhaust-nozzle areas from 1.90 to 2.51 square feet. The slope of the mean line through the data yields a value for the thrust coefficient C_F of 0.98. This thrust correlation may be used in conjunction with engine pumping characteristics to predict the jet thrust of the engine at any operating condition where sonic flow exists in the exhaust nozzle.

Combustion efficiency. - Combustion efficiency of several engines has been shown to generalize with the empirical parameter pt/V (ref. 6). However, the parameter $W_{a,2}T_9$ is used herein because it is proportional to pt/V (ref. 7) and is more convenient for calculation purposes. The variation of combustion efficiency with $W_{a,2}T_9$ is presented in figure 9 for all the Reynolds number indices and exhaust-nozzle areas included in this investigation. The combustion efficiency obtained from figure 9 can be used together with engine air flow and pumping characteristics to calculate the engine fuel requirement as shown in appendix C.

Performance Calculated from Generalized Data

Performance maps. - Over-all engine performance was calculated from generalized data for a flight Mach number of 0.7 and altitudes of 15,000, 35,000, and 50,000 feet assuming NACA standard flight conditions. These calculations included the four exhaust nozzles investigated and are presented in the form of performance maps in figure 10. Performance maps are defined by the relation between exhaust-gas total temperature and engine speed for selected values of exhaust-nozzle area, net thrust, and specific fuel consumption. These maps not only afford a convenient method of presenting a large amount of data but also show the location of specific-fuel-consumption contours. Because the total variation in specific fuel consumption was small for this engine at a given flight condition, the precise location and shape of the contours is uncertain. Although the 2.07-square-foot exhaust nozzle gave minimum specific fuel consumption, it can be seen that the use of the rated exhaust nozzle (1.97 sq ft) would result in operation very close to the minimun throughout the range of Reynolds number indices investigated.

Altitude performance. - Data for the altitude performance were calculated from the generalized performance data for the rated exhaust nozzle at altitudes from sea level to 55,000 feet, flight speeds from zero to 1100 knots, and engine speeds from 7000 to 8300 rpm. These charts (fig. 11) show the variation of net thrust with the true air speed at each altitude. Superimposed are lines of constant engine speed, fuel flow, and air flow. The highest flight speed on each chart corresponds to the flight Mach number at which the limiting compressor inlet temperature ($T_2 = 200^\circ$ F) is reached. In addition, there is a line on each chart corresponding to the engine speed at which the exhaust gas reached an average total temperature of 1166° F. The sea-level performance chart (fig. 11(a)) shows for zero ram and rated conditions that the engine used for this investigation produced approximately 7700 pounds of thrust with about 123 pounds per second of air flow and a specific fuel consumption of 0.91 pound per hour per pound of thrust.

SUMMARY OF RESULTS

The over-all performance of the J65-B3 turbojet engine was determined over a range of Reynolds number indices from 0.8 to 0.2 and exhaust-nozzle areas from 1.90 to 2.51 square feet. With the rated exhaust nozzle (1.97 sq ft), the sea-level static performance of the engine operating at rated speed was: thrust, 7700 pounds; air flow, 123 pounds per second; and specific fuel consumption, 0.91 pound per hour per pound of thrust.

The variation of specific fuel consumption with both exhaust-nozzle area and engine speed was small for a particular flight condition. The use of the rated exhaust nozzle permitted operation close to the point of minimum specific fuel consumption for a wide range of flight conditions.

At a constant corrected engine speed and exhaust-nozzle area, decreasing Reynolds number index from 0.8 to 0.2 resulted in an increase in engine total-pressure and -temperature ratios of 4 and 8 percent, respectively. Within the range of variables included in this investigation, no consistent variation of corrected air flow with either Reynolds number index or exhaust-nozzle area could be detected.

The engine was operated with the rated exhaust nozzle at a flight Mach number of 0.8 up to a facility limit encountered at 75,000 feet.

Lewis Flight Propulsion Laboratory
National Advisory Committee for Aeronautics
Cleveland, Ohio, March 9, 1955

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APPENDIX A

SYMBOLS

The following symbols are used in this report:

A	area, sq ft
C_d	flow coefficient 3631
C_F	thrust coefficient - ratio of scale jet thrust to ideal jet thrust (product of ideal mass flow and ideal effective velocity)
C_t	thermal expansion coefficient
F_j	jet thrust, lb
F_n	net thrust, lb
f	fuel-air ratio
g	acceleration due to gravity, 32.2 ft/sec^2
H_a	enthalpy, Btu/lb
M	Mach number
N	engine speed, rpm
P	total pressure, lb/sq ft abs
p	static pressure, lb/sq ft abs
R	gas constant, 53.3 ft-lb/(lb)($^{\circ}$R)
Re_i	Reynolds number index, $\delta/\varphi\sqrt{\theta}$
T	total temperature, $^{\circ}$R
t	static temperature, $^{\circ}$R
V	velocity, ft/sec or knots
W_a	air flow, lb/sec
W_f	fuel flow, lb/hr

γ ratio of specific heats

δ ratio of total pressure to NACA standard sea-level static pressure

η_b combustion efficiency

θ ratio of total temperature to NACA standard sea-level static temperature

Subscripts:

0 free stream

1 engine inlet duct

2 compressor inlet

3 compressor outlet

4 turbine inlet

5 turbine outlet

9 exhaust-nozzle inlet

cr critical

f fuel

i ideal

n exhaust nozzle

ob overboard

s scale

APPENDIX B

GENERALIZED DATA

Air flow. - Engine inlet air flow was determined from the sum of exhaust-nozzle-exit weight flow, engine fuel flow, and compressor overboard air flow: 3631

$$W_{a,2} = \frac{p_{cr} A_n C_t C_d}{\sqrt{RT_9}} \sqrt{\frac{2g\gamma}{\gamma - 1}\left[1 - \left(\frac{p_{cr}}{P_9}\right)^{\frac{\gamma-1}{\gamma}}\right]\left(\frac{P_9}{p_{cr}}\right)^{\frac{2\gamma-2}{\gamma}}} - \frac{W_f}{3600} + W_{a,ob}$$

where γ was determined from fuel-air ratio and T_9 as described in reference 8. The exhaust-nozzle flow coefficients were taken from reference 2.

Combustion efficiency. - Combustion efficiency was defined as the ratio of the actual to ideal enthalpy rise across the engine:

$$\eta_b = \frac{\left.\Delta H_a\right]_2^9 + f\left[\frac{A_m + B}{m + 1}\right]_{T_f}^9 - \frac{W_{a,ob}}{W_{a,2}} \left.\Delta H_a\right]_{ob}^9}{f \times 18,700}$$

The term $\frac{A_m + B}{m + 1}$ accounts for the difference between the enthalpy of carbon dioxide and water vapor in the burned mixture and the enthalpy of the oxygen removed from the air by their formation (ref. 9).

Scale jet thrust. - Jet thrust was determined from an algebraic summation of the forces acting on the engine. Because the bellmouth was attached to the front bulkhead instead of the engine inlet duct (fig. 2), the force due to the momentum of the inlet air was included:

$$F_{j,s} = F_d + \frac{W_{a,2}}{g} V_1 + A_{seal}(p_1 - p_{tank})$$

where F_d is the force due to the null-type balance and A_{seal} is the effective area of the inlet duct at station 1.

Performance Maps

True air speed. - True air speed was calculated from the total and static pressures and temperatures corresponding to each flight condition assuming no inlet total-pressure loss:

$$V_0 = \sqrt{\frac{2\gamma gRt_0}{\gamma - 1}\left[\left(\frac{P_2}{p_0}\right)^{\frac{\gamma-1}{\gamma}} - 1\right]}$$

<u>Pumping characteristics, air flow, and fuel flow</u>. - Engine total-pressure and -temperature ratios were determined from plots of these parameters against corrected engine speed for the four exhaust nozzles and three Reynolds number indices investigated.

Engine inlet air flow was obtained by "uncorrecting" values read from figure 5.

Fuel flow was determined from plots of corrected fuel flow against corrected engine speed, because the flight conditions selected for the performance maps closely approximated the conditions at which the data were obtained.

<u>Thrust</u>. - Jet thrust was calculated from the exhaust-nozzle pressure-drop parameter:

$$F_j = A_n C_F (1.26\ P_9 - p_0)$$

The measured exhaust-nozzle thrust coefficients tabulated as follows were used:

Exhaust-nozzle area, sq ft	Thrust coefficient
1.90	0.97
1.97	.98
2.07	.98
2.51	.98

Net thrust is defined as the change in momentum imposed on the working fluid by the engine:

$$F_n = F_j - \frac{W_{a,2}}{g} V_0$$

<u>Altitude performance</u>. - The calculation of true air speed, air flow, and engine pumping characteristics was the same for the altitude performance and performance maps.

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Fuel flow. - The fuel requirement of the engine was determined from the following relation:

$$W_f = \frac{f_i}{\eta_b} 3600 \, W_{a,2}$$

The ideal fuel-air ratio was obtained from references 9 and 10 and is presented in figure 12. Engine combustion efficiency was obtained from figure 9.

Thrust. - Jet thrust was calculated as follows: 3631

$$F_j = \frac{(W_{a,2} - W_{a,ob} + W_f)}{g} V_n + A_n(p_n - p_0)$$

This equation was solved using the method of reference 8 and an effective velocity coefficient of 0.99.

APPENDIX C

NUMERICAL EXAMPLES

To illustrate the method of obtaining engine performance from generalized data, the following numerical examples are presented:

Case I

For the case when engine speed and exhaust-nozzle area are known, sea-level static operation of the engine at rated engine speed with the rated exhaust-nozzle area was selected. The following are known:

$$N = 8300 \text{ rpm}$$

$$A_n = 1.97 \text{ sq ft}$$

$$P_2 = 2116 \text{ lb/sq ft abs}$$

$$p_0 = 2116 \text{ lb/sq ft abs}$$

$$T_2 = 519^\circ \text{ R}$$

$$t_0 = 519^\circ \text{ R}$$

The following may be calculated:

$$\sqrt{\theta_2} = 1$$

$$\delta_2 = 1$$

$$Re_i = \frac{P_2(T_2 + 216)}{5.774\ T_2^{\,2}} = 1.0$$

$$N/\sqrt{\theta} = 8300 \text{ rpm}$$

$$V_0 = M\sqrt{gR\gamma t_0} = 0$$

From figure 5:

$$\frac{W_{a,2}\sqrt{\theta}}{\delta} = 122.5 \text{ lb/sec}$$

$$W_{a,2} = 122.5 \text{ lb/sec}$$

From figure 6 the pumping characteristics at a Reynolds number index of 0.4 are:

$$P_9/P_2 = 2.315$$

$$T_9/T_2 = 3.22$$

The pumping characteristics can now be adjusted for Reynolds number effects using figure 7:

$$\frac{(P_9/P_2)_{Re_i=1}}{(P_9/P_2)_{Re_i=0.4}} = 0.99$$

$$\frac{(T_9/T_2)_{Re_i=1.0}}{(T_9/T_2)_{Re_i=0.4}} = 0.976$$

Then:

$$(P_9/P_2)_{Re_i=1} = 2.315 \times 0.99 = 2.29$$

$$(T_9/T_2)_{Re_i=1} = 3.22 \times 0.976 = 3.14$$

and

$$P_9 = 2.29 \times 2116 = 4846 \text{ lb/sq ft abs}$$

$$T_9 = 3.14 \times 519 = 1629^{\circ} \text{ R}$$

Jet thrust can be obtained from figure 8 as follows:

$$\frac{F_j}{A_n C_F} = 1.26\ P_9 - p_0$$

$$F_j = 1.97 \times 0.98(1.26 \times 4846 - 2116)$$

$$F_j = 7700 \text{ lb}$$

Net thrust:

$$F_n = F_j - \frac{W_{a,2}}{g} V_0 = 7700 \text{ lb}$$

To calculate the fuel requirement of the engine, the following steps are necessary:

The ideal fuel-air ratio from figure 12 is $f_i = 0.0157$

From figure 9, the combustion efficiency is $\eta_b = 0.99$

Dividing the ideal fuel-air ratio by combustion efficiency to obtain the actual fuel-air ratio gives:

$$f = \frac{0.0157}{0.099} = 0.0159$$

Then:

$$W_f = f \times W_{a,2} \times 3600$$

$$W_f = 0.0159 \times 122.5 \times 3600 = 7012 \text{ lb/hr}$$

The specific fuel consumption can then be determined as

$$sfc = \frac{W_f}{F_n} = \frac{7012}{7700} = 0.91$$

Case II

For the case when engine speed and turbine-discharge temperature are known, the calculation procedure is identical to Case I except for the method of determining pumping characteristics. To illustrate this difference, the following conditions were selected:

$$P_2 = 1656 \text{ lb/sq ft abs}$$

$$T_2 = 511^{\circ} \text{ R}$$

$$T_9 = 1625^{\circ} \text{ R}$$

$$N = 8300 \text{ rpm}$$

The following may be calculated:

$$N/\sqrt{\theta} = 8350 \text{ rpm}$$

$$T_9/T_2 = 3.18$$

$$Re_i = 0.8$$

Engine temperature ratio can be adjusted to a Reynolds number index of 0.4 using figure 7:

$$\left(\frac{T_9}{T_2}\right)_{Re_i=0.4} = \frac{3.18}{0.976} = 3.26$$

Entering figure 6 with corrected engine speed and the adjusted temperature ratio gives: 3637

$$(P_9/P_2)_{Re_i=0.4} = 2.335$$

$$A_n = 1.97$$

The engine total-pressure ratio can be adjusted for Reynolds number effects using figure 7:

$$(P_9/P_2)_{Re_i=0.8} = 2.335 \times 0.99 = 2.32$$

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TABLE I. - COMPONENT PERFORMANCE OF J65-B3 TURBOJET ENGINE

Engine-inlet Reynolds number index, Re_1	Inlet total pressure, P_2, lb/sq ft abs	Inlet total temperature, T_2, °R	Engine-exhaust-ambient pressure, p_0, lb/sq ft abs	Compressor-outlet total pressure, P_3, lb/sq ft abs	Compressor-outlet total temperature, T_3, °R	Turbine-inlet total pressure, P_4, lb/sq ft abs	Turbine-inlet total temperature, T_4, °R	Turbine-outlet total pressure, P_5, lb/sq ft abs	Exhaust-nozzle inlet total temperature, T_9, °R	Exhaust-nozzle inlet total pressure, P_9, lb/sq ft abs	Engine-inlet air flow, $W_{a,2}$, lb/sec	Overboard air flow, $W_{a,ob}$, lb/sec	Combustor efficiency, η_b	Fuel flow, W_f, lb/hr	Engine speed, N, rpm	Scale jet thrust $F_{j,s}$, lb
Exhaust nozzle area, 1.90 sq ft																
0.796	1762	537	801	4,858	765	4,646	1087	1748	861	1687	56.27	0.90	0.955	880	5978	2395
.798	1766	537	803	5,895	802	5,631	1258	2121	1001	2048	63.21	.99	.977	1415	6357	3242
.795	1758	537	808	6,028	806	5,760	1277	2158	1016	2087	63.95	1.02	.975	1483	6387	3416
.796	1729	530	781	6,637	820	6,346	1377	2387	1100	2309	67.78	1.00	.989	1856	6597	3938
.796	1761	537	804	7,241	846	6,934	1460	2605	1167	2541	72.49	1.11	.997	2185	6779	4349
.802	1729	527	787	7,312	838	6,999	1464	2622	1170	2543	72.21	1.02	.977	2221	6796	4416
.793	1754	537	807	8,520	888	8,181	1657	3080	1334	3011	80.00	1.21	.997	3101	7170	5461
.797	1735	531	804	9,214	906	8,851	1778	3337	1437	3264	83.08	1.22	.997	3704	7394	6118
.792	1715	529	786	10,437	944	10,040	1960	3806	1593	3721	89.93	1.23	1.002	4744	7786	7174
.792	1717	529	787	10,877	956	10,462	2008	3950	1631	3861	92.17	1.24	.998	5079	7911	7476
.399	625	413	307	2,529	653	2,424	1140	915	905	885	28.66	.39	.973	670	5925	1441
.394	621	412	310	2,732	668	2,621	1211	985	962	952	29.89	.42	.969	792	6053	1551
.402	623	413	313	2,905	684	2,786	1283	1051	1023	1019	30.98	.42	1.007	907	6209	1823
.399	626	413	311	3,351	714	3,213	1416	1211	1132	1174	33.83	.45	.987	1184	6517	2128
.398	626	414	310	3,630	734	3,491	1498	1309	1201	1275	35.58	.45	.997	1385	6708	2368
.398	626	414	308	3,987	768	3,834	1619	1438	1295	1402	37.63	.47	.995	1625	6978	2671
.399	627	414	305	4,103	770	3,946	1659	1495	1335	1454	38.40	.46	1.011	1707	7093	2820
.395	615	411	307	4,342	785	4,187	1735	1568	1397	1531	39.54	.52	.973	1991	7298	2981
.397	624	414	309	4,644	809	4,484	1848	1690	1497	1645	40.91	.50	.998	2177	7548	3238
.395	617	413	304	4,716	816	4,533	1888	1721	1532	1680	41.30	.51	1.000	2279	7637	3331
.393	621	416	310	4,918	835	4,757	1967	1797	1601	1752	42.15	.53	.991	2475	7795	3490
.193	300	411	138	1,543	712	1,486	1402	556	1117	538	15.59	.19	.968	541	6323	966
.188	299	411	138	1,631	716	1,571	1455	586	1169	568	16.09	.20	.966	602	6442	999
.191	297	411	138	1,770	740	1,703	1567	638	1264	619	16.79	.20	.958	720	6694	1152
.190	294	410	138	1,912	757	1,838	1642	688	1325	669	17.70	.21	.941	834	6864	1255
.187	301	410	140	2,130	784	2,047	1770	773	1434	752	19.12	.22	.997	991	7126	1395
.192	302	409	138	2,253	798	2,168	1849	745	1504	792	19.65	.23	.948	1105	7323	1551
.192	302	411	137	2,279	814	2,201	1914	826	1558	804	19.56	.23	.963	1165	7491	1612
.194	307	413	137	2,391	821	2,231	1944	864	1586	841	20.28	.23	.979	1230	7548	1669
.194	305	415	135	2,443	831	2,362	1981	888	1618	864	20.62	.24	1.000	1280	7617	1726
Exhaust nozzle area, 1.97 sq ft																
0.797	1748	534	800	5,301	779	5,078	1111	1813	869	1744	60.08	0.92	0.951	991	6205	2597
.800	1755	534	795	6,198	808	5,928	1229	2106	961	2021	66.50	1.00	----	1421	6507	3393
.797	1749	534	805	6,763	827	6,473	1321	2294	1039	2210	69.91	1.04	.970	1725	6696	3788
.795	1747	535	800	7,441	849	7,119	1414	2530	1114	2440	74.40	1.09	.978	2102	6886	4379
.800	1759	535	805	8,469	883	8,119	1567	2883	1241	2805	80.83	1.16	.992	2770	7239	5282
.799	1753	534	793	9,155	906	8,793	1665	3119	1323	3042	84.55	1.18	.975	3253	7440	5911
.796	1746	534	804	10,234	943	9,839	1826	3486	1457	3399	89.98	1.26	.996	4138	7807	6829
.802	1755	533	790	11,521	978	11,093	2001	3949	1605	3835	96.52	1.33	.987	5230	8199	7989
.798	1738	531	789	11,507	978	11,055	2000	3954	1604	3849	96.91	1.32	.986	5259	8239	8003
.400	612	406	515	2,647	656	2,540	1150	926	905	894	30.23	.40	.969	725	6024	1177
.402	612	404	509	3,010	680	2,894	1242	1028	979	998	32.41	.44	.959	907	6300	1474
.401	606	402	501	3,066	682	2,944	1260	1032	990	1003	32.38	.43	.948	938	6349	1510
.395	599	403	498	3,322	703	3,191	1345	1112	1059	1082	33.74	.45	.958	1093	6513	1740
.397	604	404	500	3,566	725	3,422	1424	1199	1132	1169	35.19	.46	----	1285	6795	1943
.395	604	406	495	3,843	746	3,695	1516	1313	1200	1282	37.41	.47	.978	1454	7010	2135
.396	637	429	541	4,461	808	4,290	1708	1500	1364	1463	39.97	.54	.973	1862	7470	2541
.401	626	412	530	4,354	785	4,201	1660	1475	1319	1433	39.78	.48	.987	1766	7387	2528
.390	641	428	542	4,708	835	4,542	1809	1606	1444	1563	41.37	.48	.997	2064	7753	2800
.395	624	416	523	5,170	873	5,004	2016	1795	1619	1746	43.50	.53	.970	2686	8278	3335
.195	309	416	176	1,720	735	1,655	1444	580	1144	563	17.01	.21	.974	607	6653	1030
.194	309	419	147	1,729	736	1,662	1452	585	1155	567	16.90	.21	.969	612	6675	1113
.197	309	413	154	1,890	752	1,818	1534	636	1218	619	17.93	.28	.963	720	6854	1206
.201	315	413	150	2,068	775	1,987	1635	700	1309	683	19.04	.24	.958	860	7118	1396
.199	311	413	145	2,304	809	2,223	1798	789	1446	770	20.38	.25	.967	1065	7495	1610
.194	304	413	151	2,431	839	2,347	1937	829	1561	809	20.57	.27	.959	1218	7815	1654

[redacted]

TABLE I. - Concluded. COMPONENT PERFORMANCE OF J65-B3 TURBOJET ENGINE

Engine-inlet Reynolds number index, Re_1	Inlet total pressure, P_2, lb/sq ft abs	Inlet total temperature, T_2, °R	Engine-exhaust-ambient pressure, p_0, lb/sq ft abs	Compressor-outlet total pressure, P_3, lb/sq ft abs	Compressor-outlet total temperature, T_3, °R	Turbine-inlet total pressure, P_4, lb/sq ft abs	Turbine-inlet total temperature, T_4, °R	Turbine-outlet total pressure, P_5, lb/sq ft abs	Exhaust-nozzle inlet total temperature, T_9, °R	Exhaust-nozzle inlet total pressure, P_9, lb/sq ft abs	Engine-inlet air flow, $W_{a,2}$, lb/sec	Overboard air flow, $W_{a,ob}$, lb/sec	Combustor efficiency, η_b	Fuel flow, W_f, lb/hr	Engine speed, N, rpm	Scale jet thrust, $F_{j,s}$, lb
Exhaust nozzle area, 2.07 sq ft																
0.796	1741	533	796	6,274	813	5,969	1215	2025	940	1937	68.08	0.96	0.975	1337	6610	3340
.794	1738	533	797	7,478	851	7,139	1376	2406	1071	2314	76.02	1.05	.993	1959	7004	4273
.797	1741	532	796	8,693	890	8,315	1538	2779	1202	2688	83.17	1.12	.999	2679	7404	5280
.798	1743	532	802	9,886	929	9,469	1696	3178	1336	3083	90.03	1.20	.985	3553	7799	6289
.797	1735	531	798	10,408	948	9,984	1787	3373	1407	3268	92.90	1.22	.991	4018	7996	6834
.797	1736	530	809	11,265	973	11,122	1913	3681	1516	3556	97.31	1.33	1.00	4732	8290	7507
.395	613	410	303	2,585	655	2,478	1056	820	822	786	29.52	.39	.964	588	6057	1367
.400	619	409	308	2,853	674	2,734	1143	912	885	880	31.82	.42	.963	735	6252	1600
.396	610	408	305	3,129	693	3,000	1229	1001	953	969	33.72	.44	.965	894	6481	1810
.398	613	408	306	3,286	704	3,154	1276	1040	991	1007	34.35	.45	.955	987	6619	1974
.397	635	420	321	3,645	744	3,496	1377	1173	1070	1131	37.04	.44	.967	1180	6913	2236
.401	640	419	325	3,801	751	3,643	1419	1229	1106	1188	38.23	.46	.971	1285	7010	2339
.393	604	407	303	3,793	745	3,640	1445	1217	1129	1181	37.62	.47	.969	1333	7071	2422
.401	638	418	317	4,163	775	3,996	1525	1339	1193	1298	40.12	.48	.978	1522	7292	2666
.395	610	409	308	4,067	761	3,905	1509	1313	1182	1275	39.61	.49	.972	1505	7316	2623
.398	613	408	305	4,251	777	4,085	1574	1372	1234	1331	40.42	.49	.983	1630	7424	2784
.397	614	409	308	4,478	800	4,313	1666	1448	1310	1403	41.30	.49	.985	1825	7680	2931
.398	621	419	311	4,725	842	4,560	1797	1539	1419	1489	42.03	.48	.988	2075	8000	3194
.397	618	411	304	4,755	833	4,587	1802	1548	1424	1498	42.31	.49	.987	2120	7984	3221
.193	323	430	160	1,497	711	1,439	1235	485	963	465	16.07	.19	.972	415	6369	817
.194	320	429	158	1,706	743	1,641	1366	545	1068	526	17.24	.20	.958	545	6692	995
.194	309	418	162	2,062	783	1,981	1582	666	1245	646	19.50	.24	.968	802	7241	1297
.194	308	418	163	2,254	814	2,171	1714	735	1356	714	20.40	.24	.958	970	7596	1416
.195	311	419	157	2,406	851	2,320	1850	785	1465	761	21.12	.25	.967	1120	7886	1598
.195	312	420	154	2,523	865	2,436	1937	826	1542	800	21.61	.25	.966	1240	8097	1704
.195	314	422	155	2,608	890	2,514	----	859	1636	831	21.73	.25	.973	1360	8312	1778
Exhaust nozzle area, 2.51 sq ft																
0.797	1728	529	784	6,420	817	6,100	1130	1776	847	1605	72.50	0.92	0.976	1109	6796	3007
.798	1729	529	787	7,502	853	7,156	1267	2076	954	1869	79.40	.99	.985	1602	7193	3875
.794	1721	529	787	8,588	889	8,185	1396	2382	1054	2156	86.96	1.06	.999	2177	7593	4656
.797	1726	529	788	9,375	918	8,948	1500	2621	1135	2372	92.09	1.12	1.006	2658	7901	5395
.793	1718	529	795	9,905	937	9,471	1568	2764	1188	2501	94.85	1.15	1.005	2995	8096	5761
.791	1722	531	790	10,358	956	9,908	1634	2889	1240	2619	97.16	1.17	.998	3340	8293	6225
.397	623	414	312	2,328	646	2,219	918	651	684	592	29.79	.39	.995	375	5993	1024
.403	631	415	313	2,851	683	2,727	1047	792	780	718	33.74	.43	.976	588	6417	1412
.398	629	416	309	3,293	717	3,141	1164	908	869	825	36.69	.45	.978	793	6796	1747
.401	618	408	304	3,295	708	3,145	1159	910	866	826	35.78	.44	.944	812	6810	1817
.398	634	418	316	3,723	751	3,559	1283	1025	962	931	39.28	.48	.973	1034	7194	2146
.398	634	418	325	4,114	787	3,935	1408	1142	1057	1038	41.70	.48	.983	1285	7601	2447
.399	638	420	308	4,382	816	4,195	1511	1222	1141	1108	42.64	.49	.992	1488	7892	2716
.398	633	418	322	4,588	836	4,403	1596	1293	1209	1168	43.73	.51	.991	1672	8100	2811
.397	631	418	313	4,753	858	4,567	1686	1348	1283	1210	43.96	.53	.992	1849	8291	2996
.196	306	413	148	1,102	647	1,054	----	304	704	275	13.90	.19	----	----	5915	508
.199	310	412	140	1,399	687	1,340	1100	391	819	352	16.14	.21	.955	328	6401	767
.193	299	410	151	1,406	689	1,347	1105	392	830	355	16.17	.21	.955	332	6436	694
.191	295	409	141	1,640	724	1,572	1231	457	925	416	17.91	.20	.994	437	6829	903
.198	307	410	139	1,678	726	1,607	1240	468	935	420	17.98	.23	.965	459	6852	952
.197	305	409	148	1,864	755	1,784	1356	530	1025	475	19.40	.24	.982	574	7181	1083
.196	303	409	137	2,097	795	2,007	1502	588	1142	525	20.26	.24	.960	741	7643	1286
.193	299	409	143	2,200	818	2,111	1591	626	1213	558	20.85	.25	.960	838	7895	1367
.192	296	408	148	2,262	836	2,172	1661	646	1270	578	21.06	.24	.970	902	8071	1386
.194	300	409	143	2,282	841	2,191	1680	654	1286	581	21.03	.25	.960	927	8117	1450
.194	300	409	143	2,336	856	2,244	1740	670	1335	595	21.07	.20	.955	992	8263	1439

[redacted]

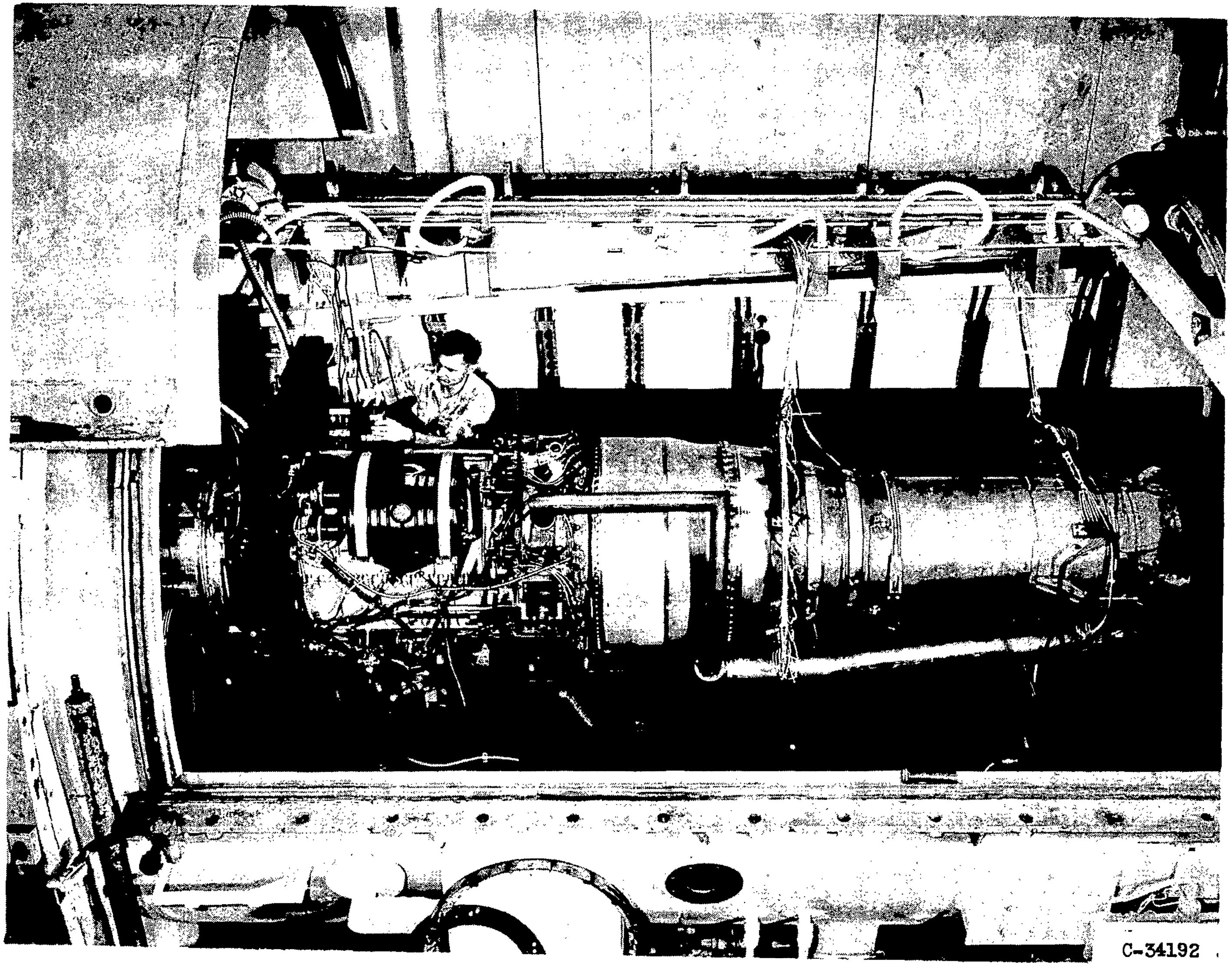
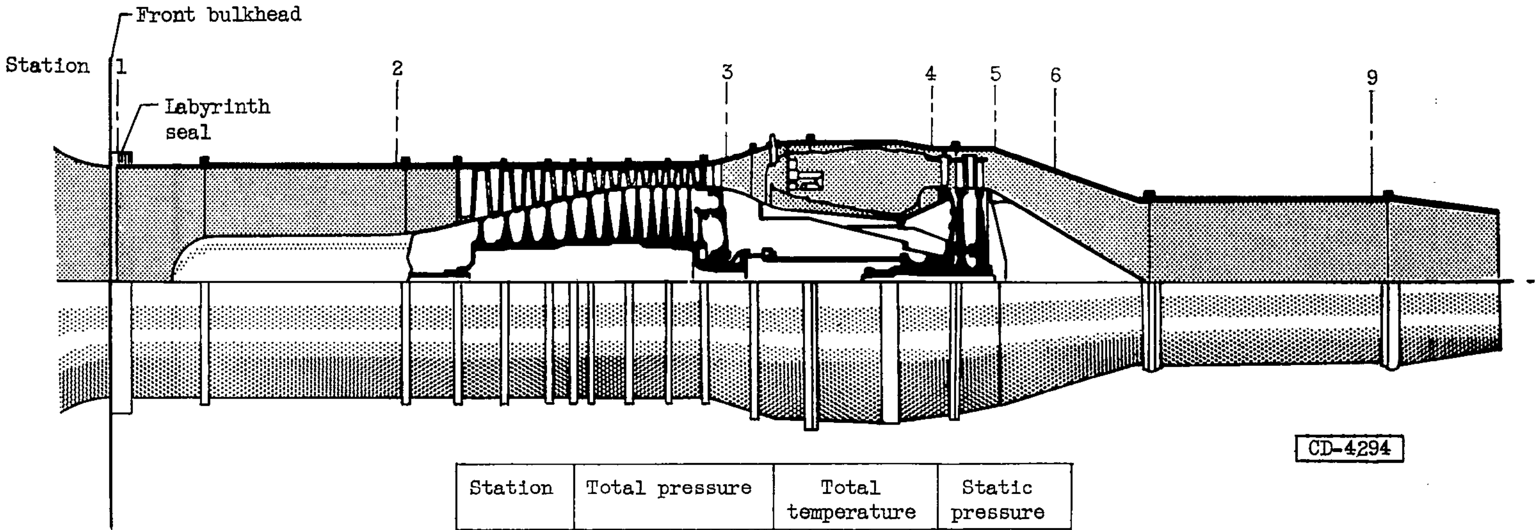


Figure 1. - J65-B3 Turbojet engine in altitude test chamber.



Station	Total pressure	Total temperature	Static pressure
2	20	12	8
3	12	12	-
4	4	-	-
5	15	-	-
6	-	4[a]	-
9	35	30	5

a - Manufacturers instrumentation.

Figure 2. - Schematic diagram of engine showing instrumentation stations.

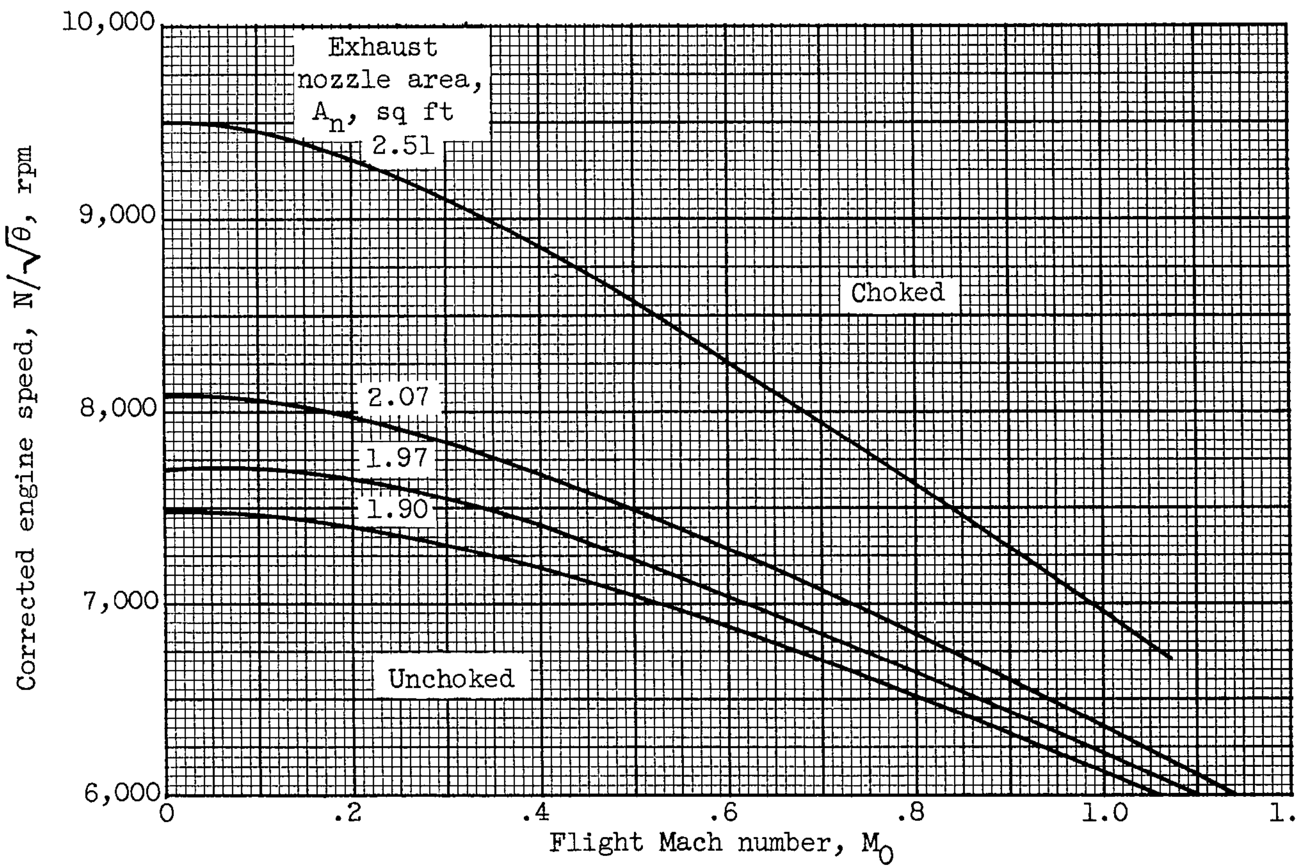


Figure 3. - Minimum corrected engine speed at which exhaust nozzle may be considered fully choked.

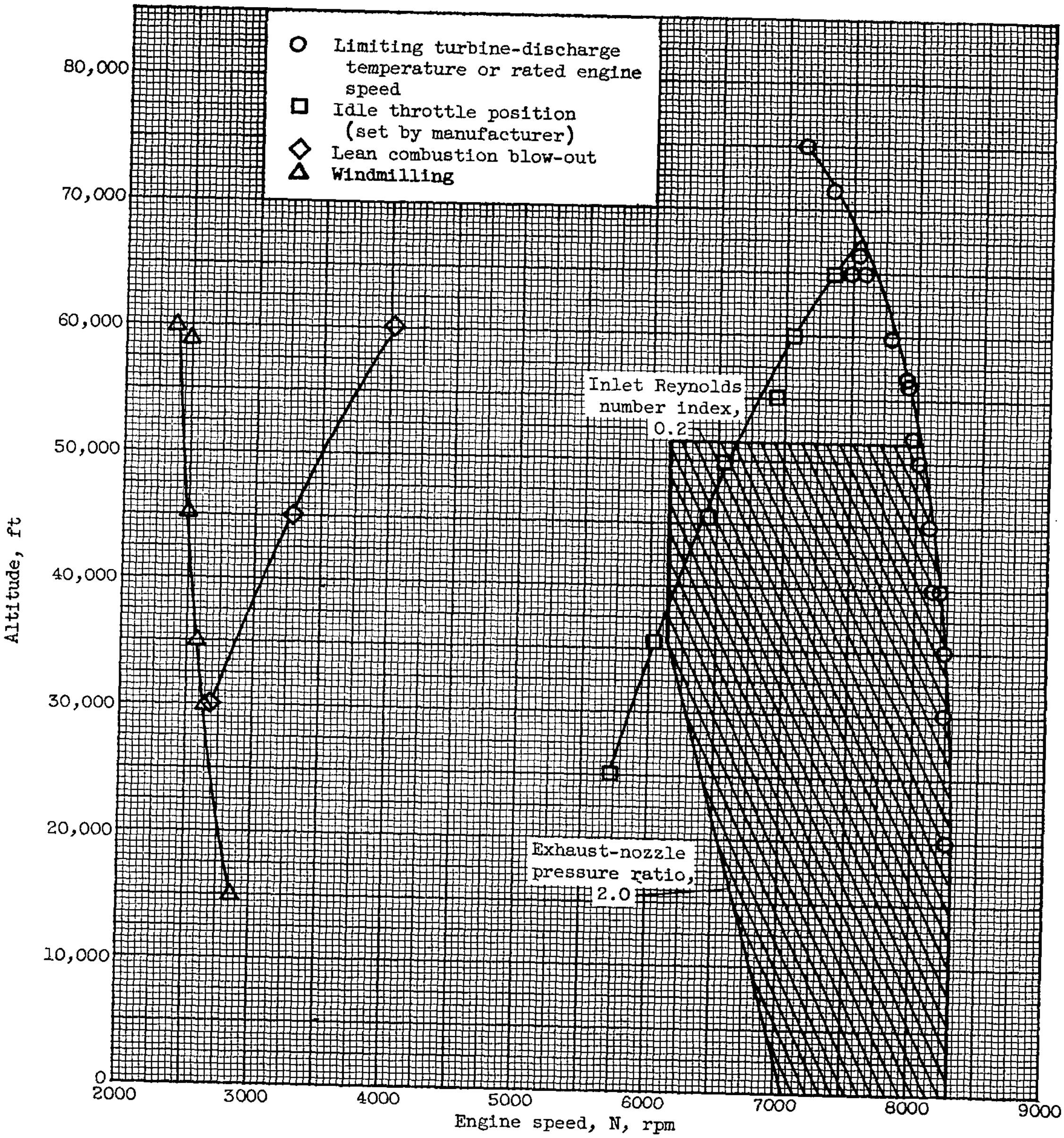


Figure 4. - Effect of altitude on engine operational limits. Flight Mach number, 0.8; rated exhaust nozzle (1.97 sq ft).

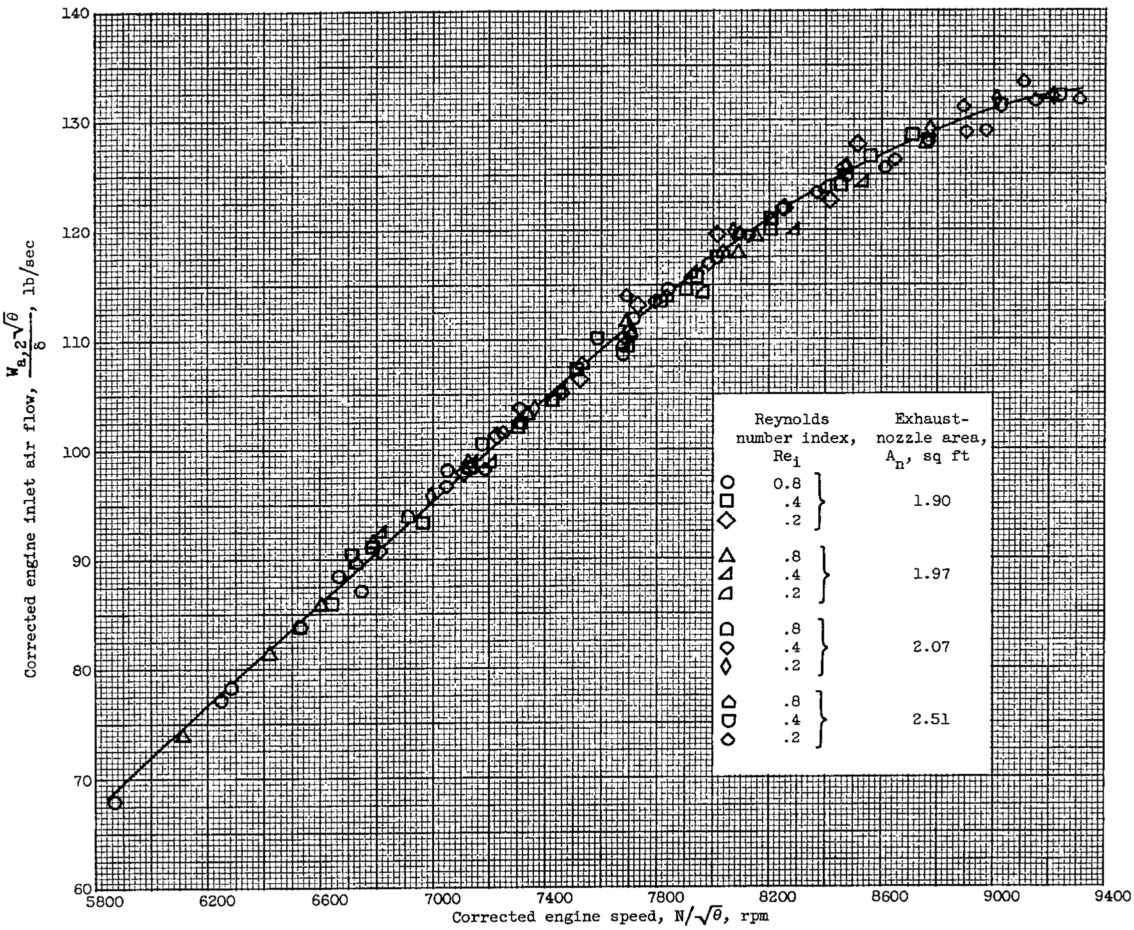


Figure 5. - Variation of corrected air flow with corrected engine speed for Reynolds number indices from 0.8 to 0.2 and exhaust-nozzle areas from 1.90 to 2.51 square feet.

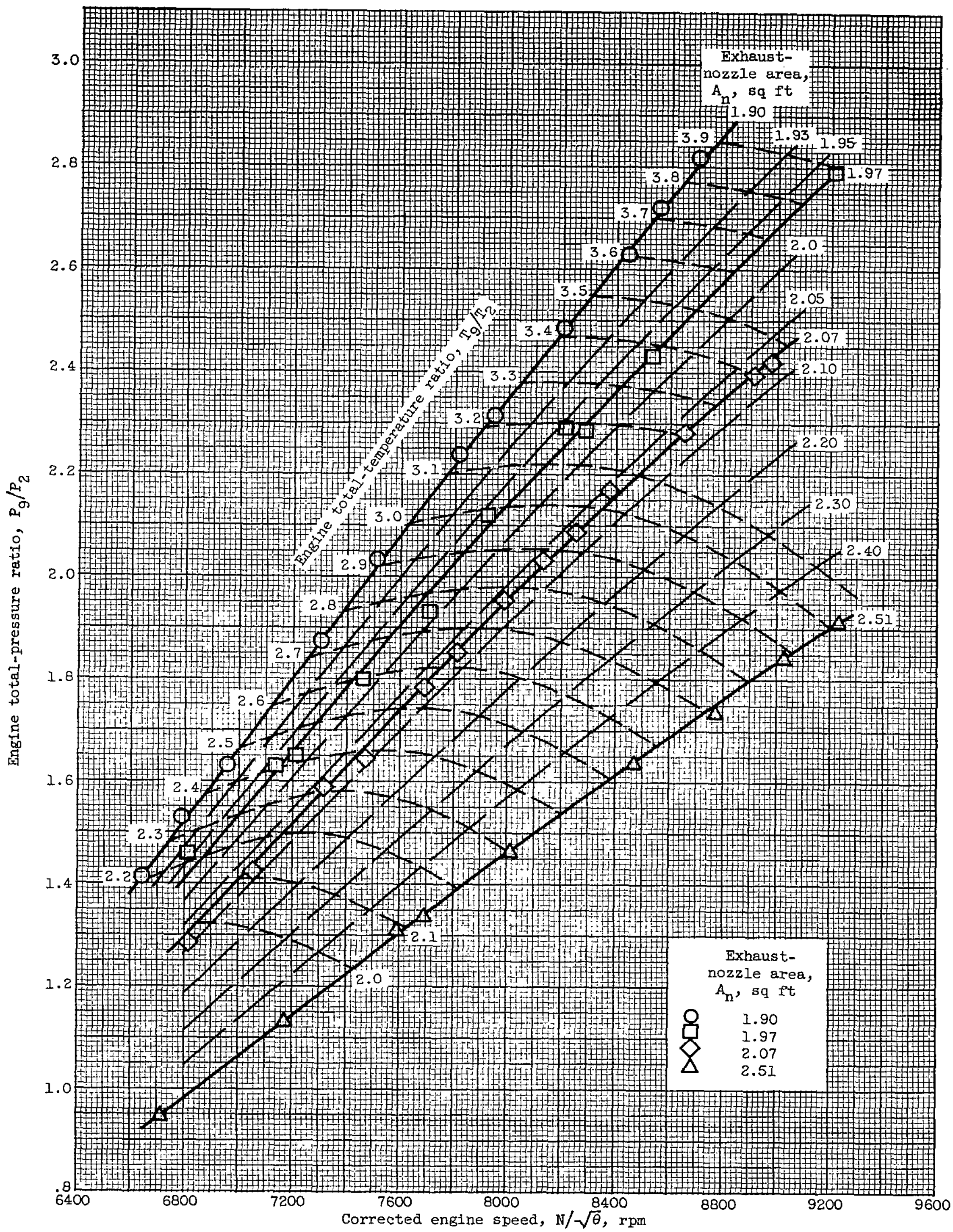


Figure 6. - Engine pumping characteristics at an inlet Reynolds number index of 0.4.

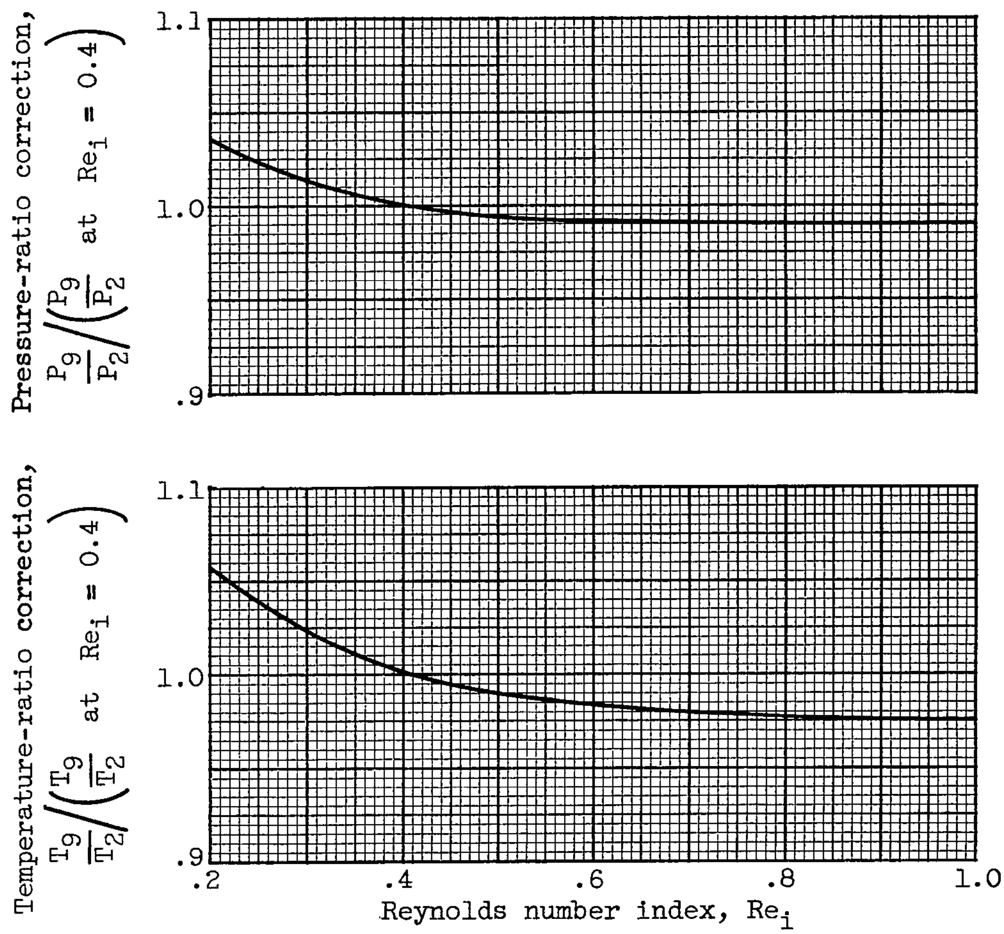


Figure 7. - Effect of Reynolds number index on engine pumping characteristics.

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CN-4 back

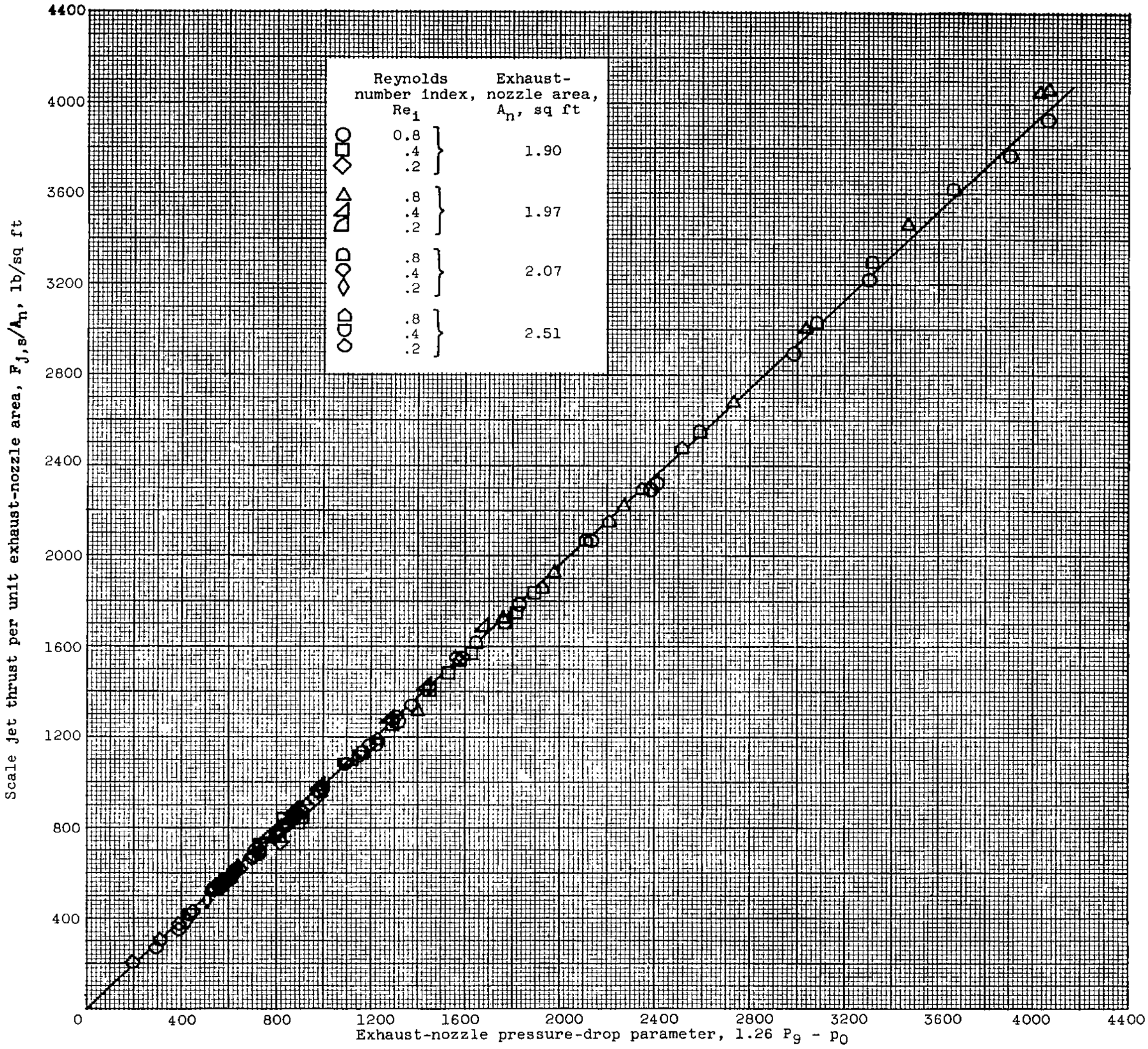


Figure 8. - Correlation of jet thrust for Reynolds number indices from 0.8 to 0.2 and exhaust-nozzle areas from 1.90 to 2.51 square feet.

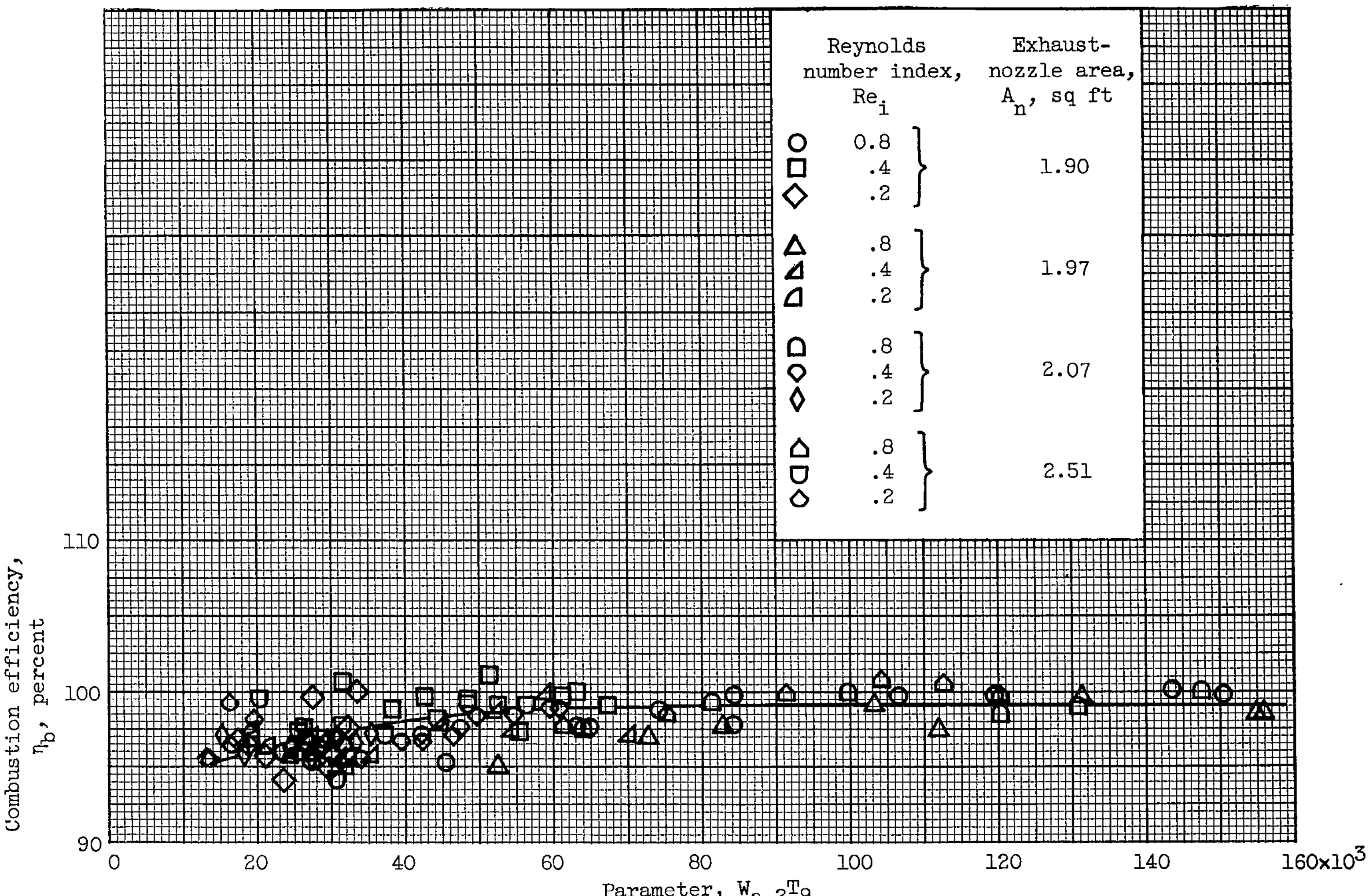
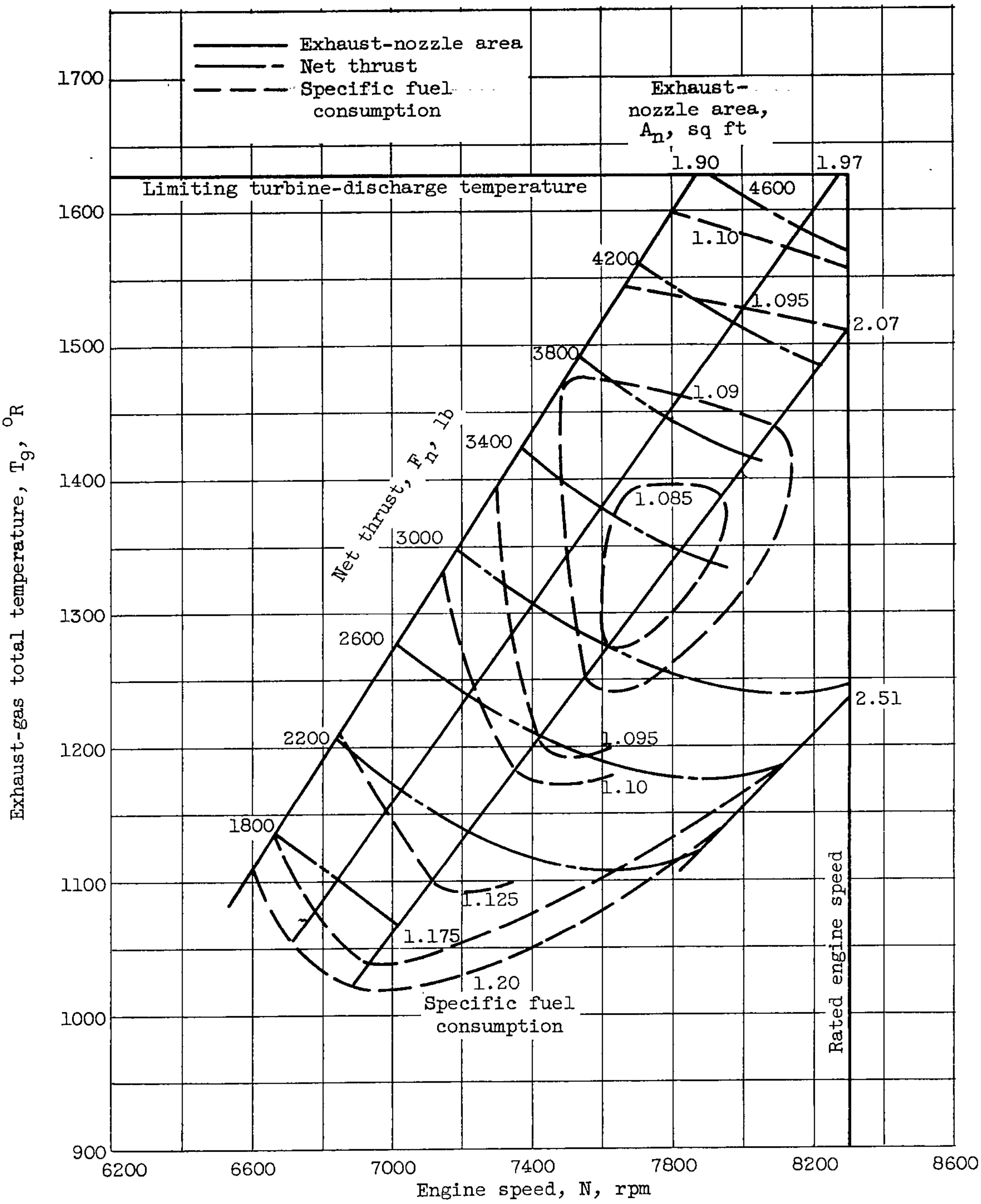
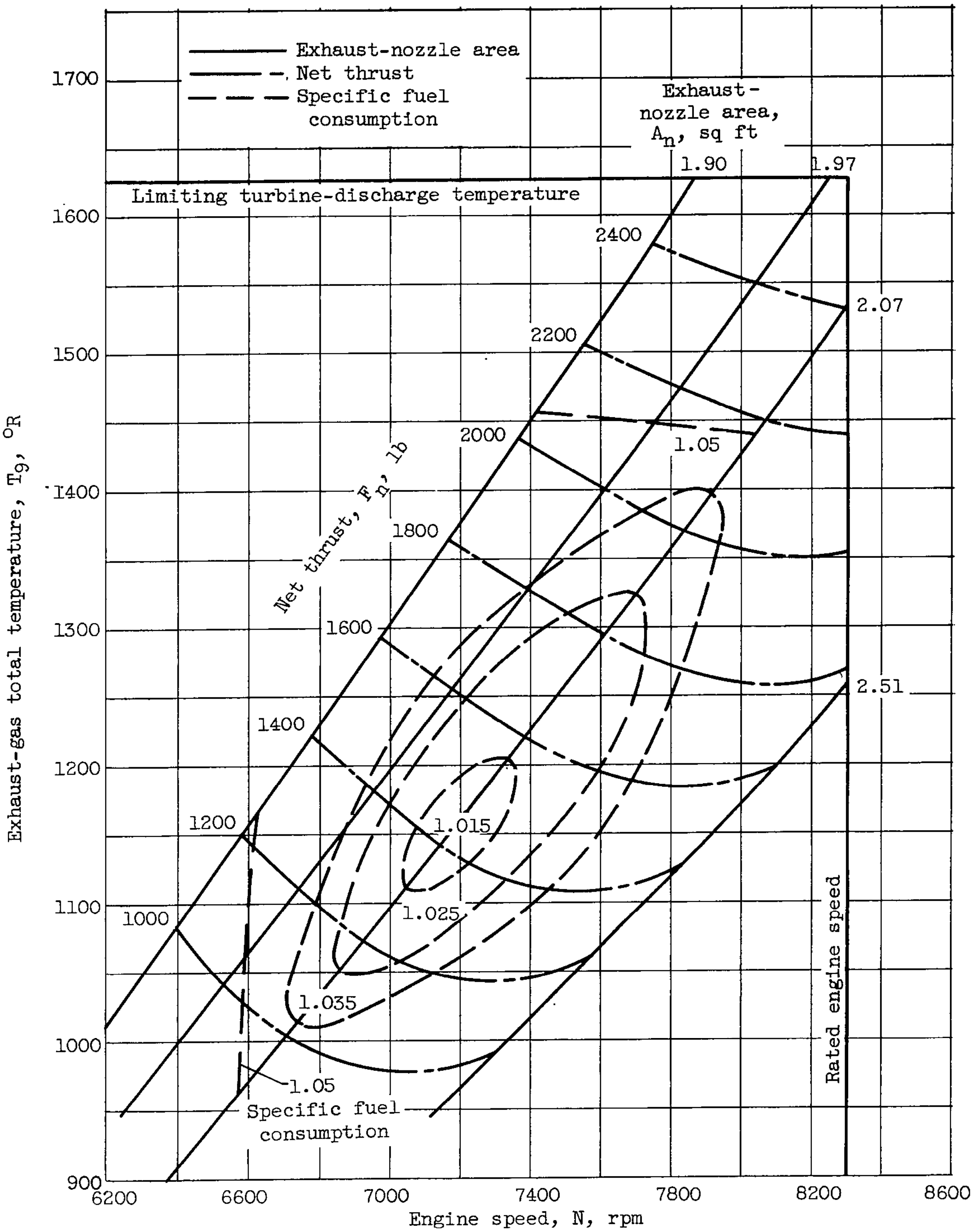


Figure 9. - Correlation of combustion efficiency for Reynolds number indices from 0.8 to 0.2 and exhaust-nozzle areas from 1.90 to 2.51 square feet.



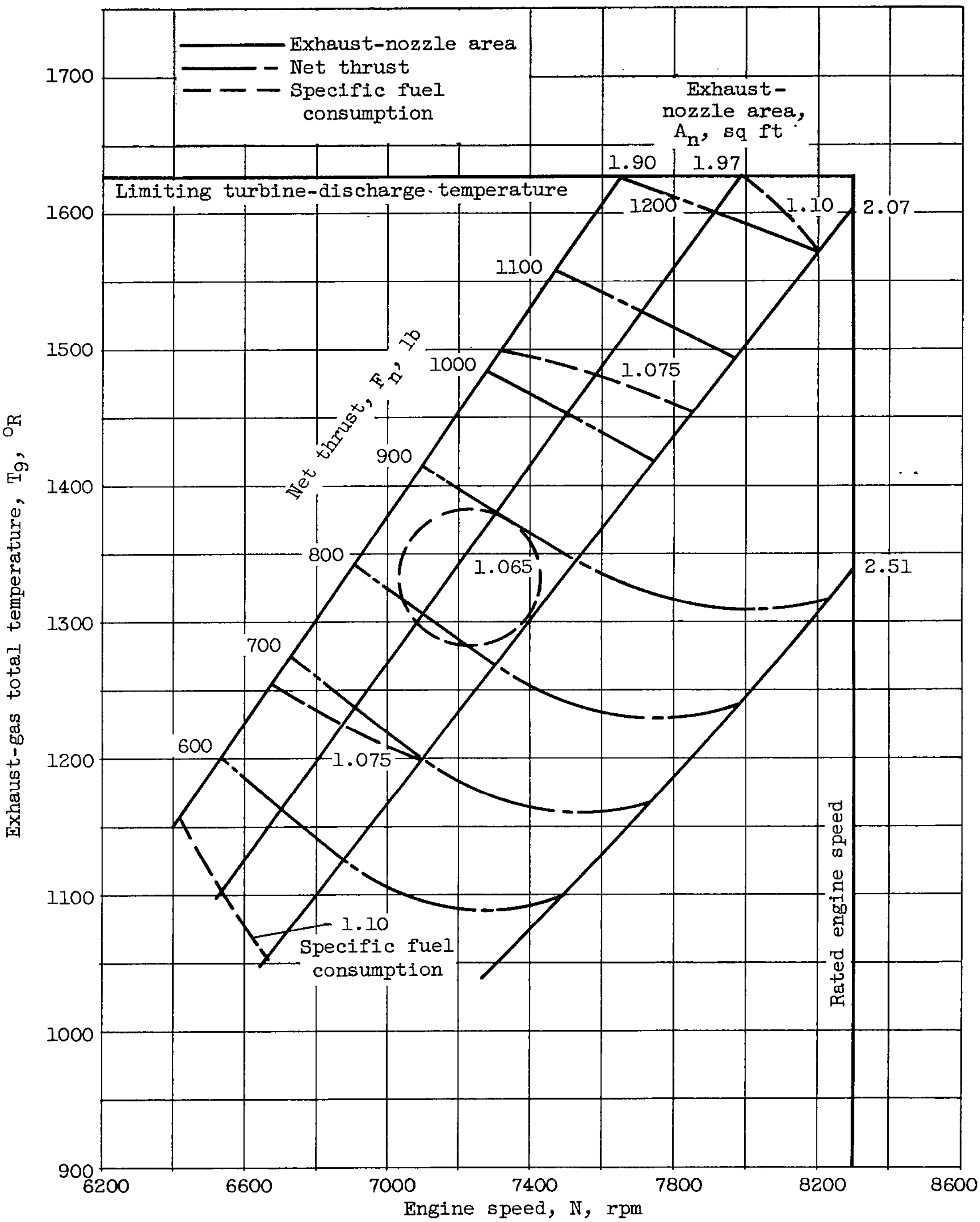
(a) Altitude, 15,000 feet.

Figure 10. - Engine performance maps. Flight Mach number, 0.7.



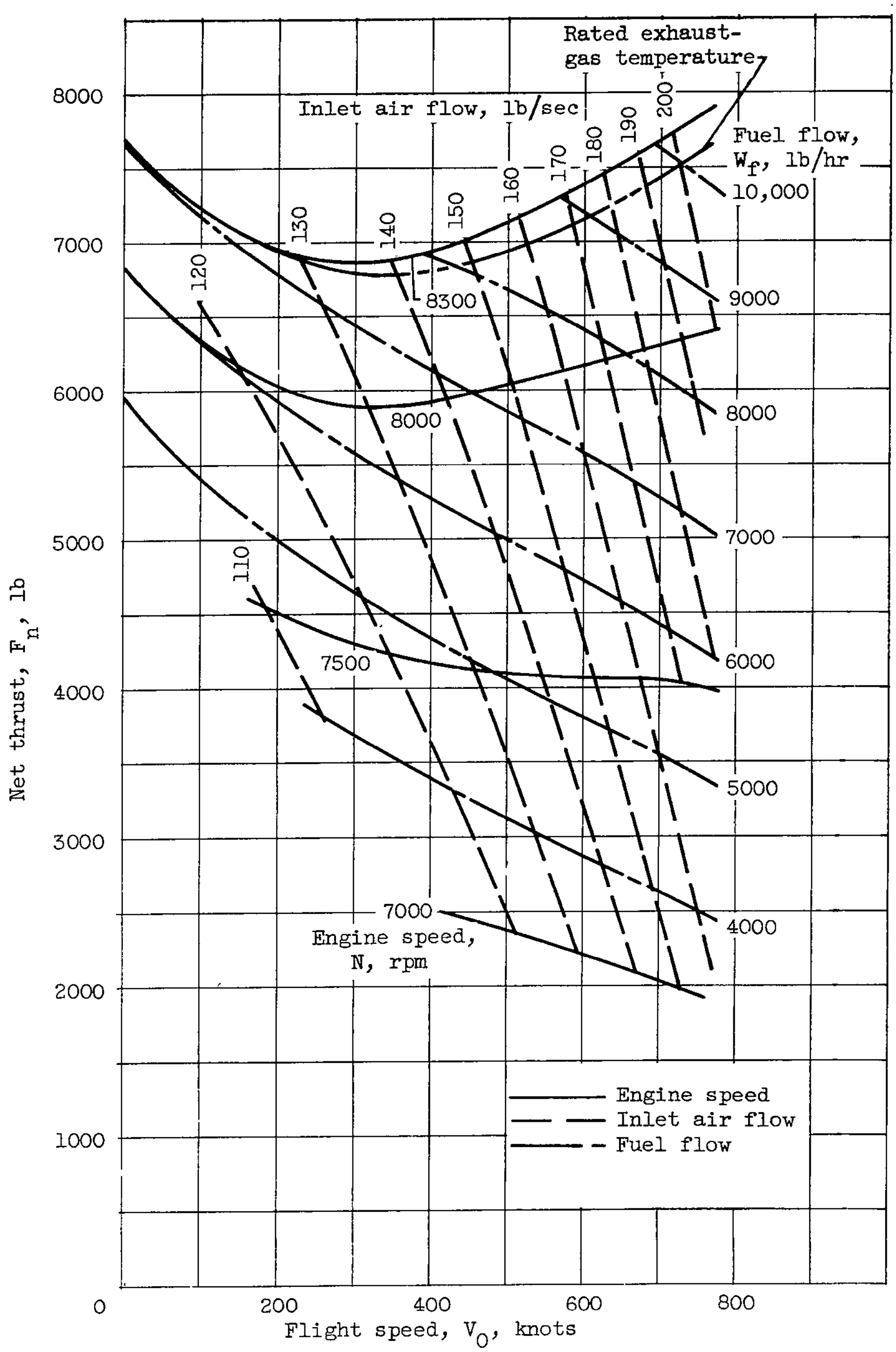
(b) Altitude, 35,000 feet.

Figure 10. - Continued. Engine performance maps. Flight Mach number, 0.7.



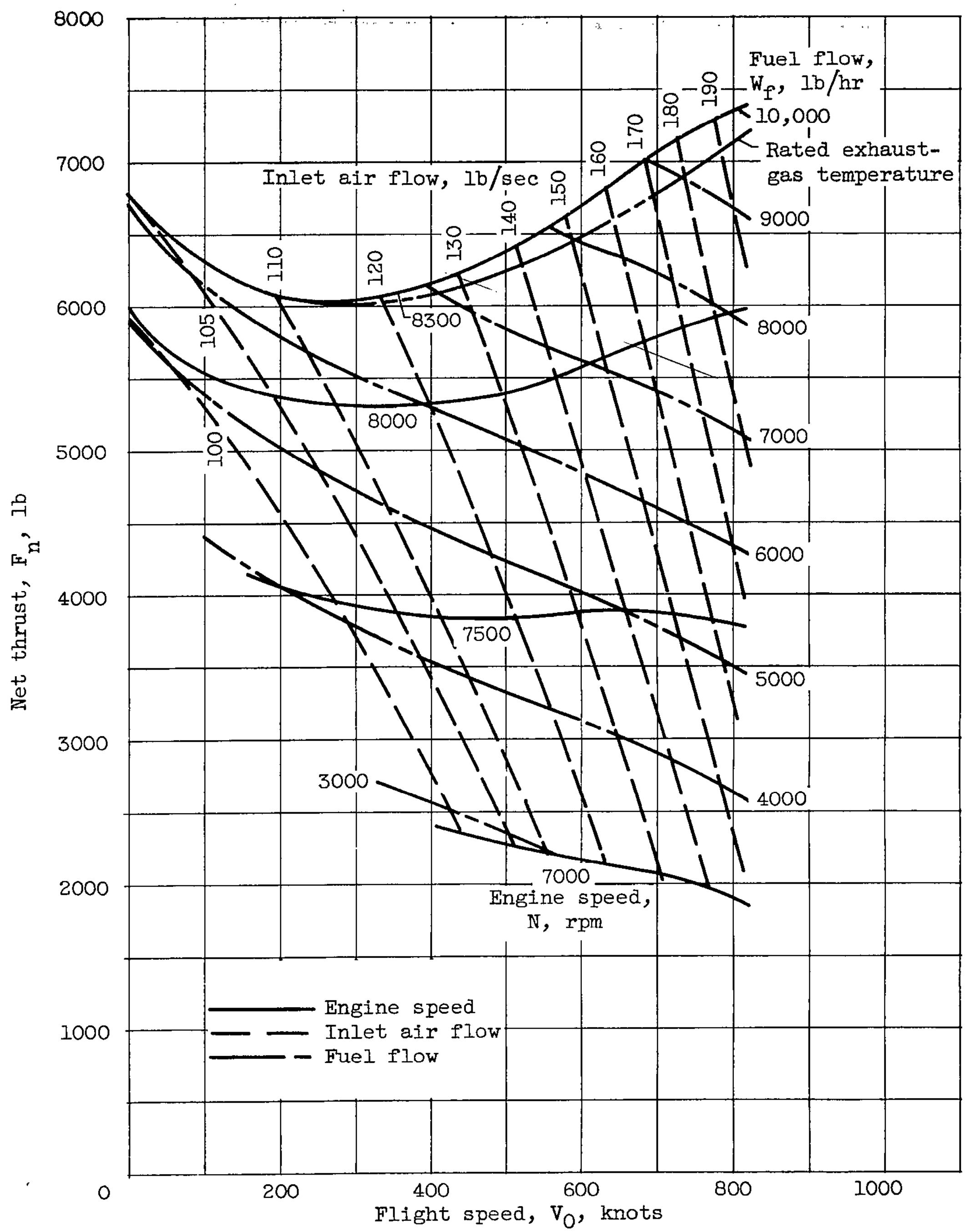
(c) Altitude, 50,000 feet.

Figure 10. - Concluded. Engine performance maps. Flight Mach number, 0.7.



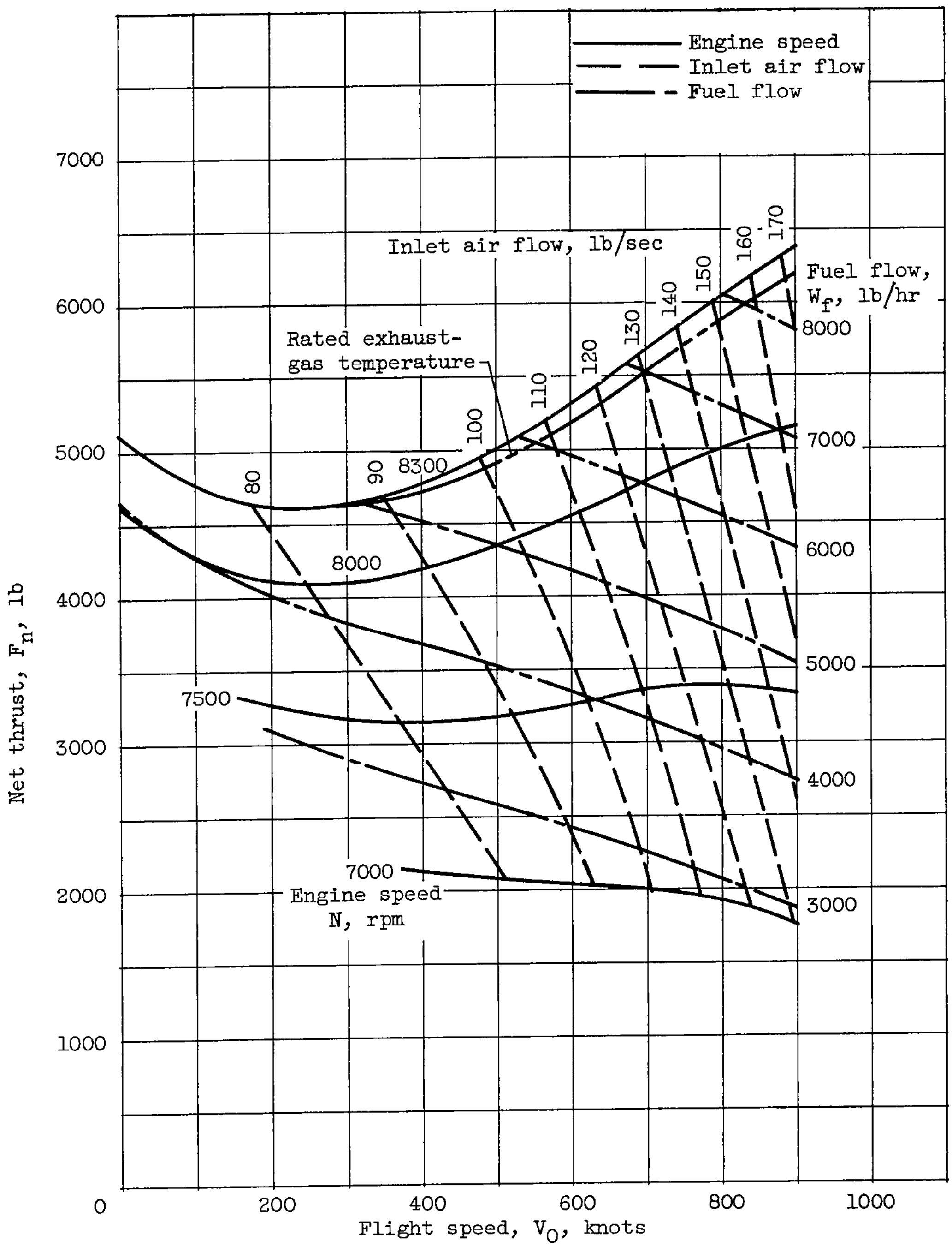
(a) Altitude, sea level.

Figure 11. - Altitude performance calculated from pumping characteristics for rated exhaust-nozzle area.



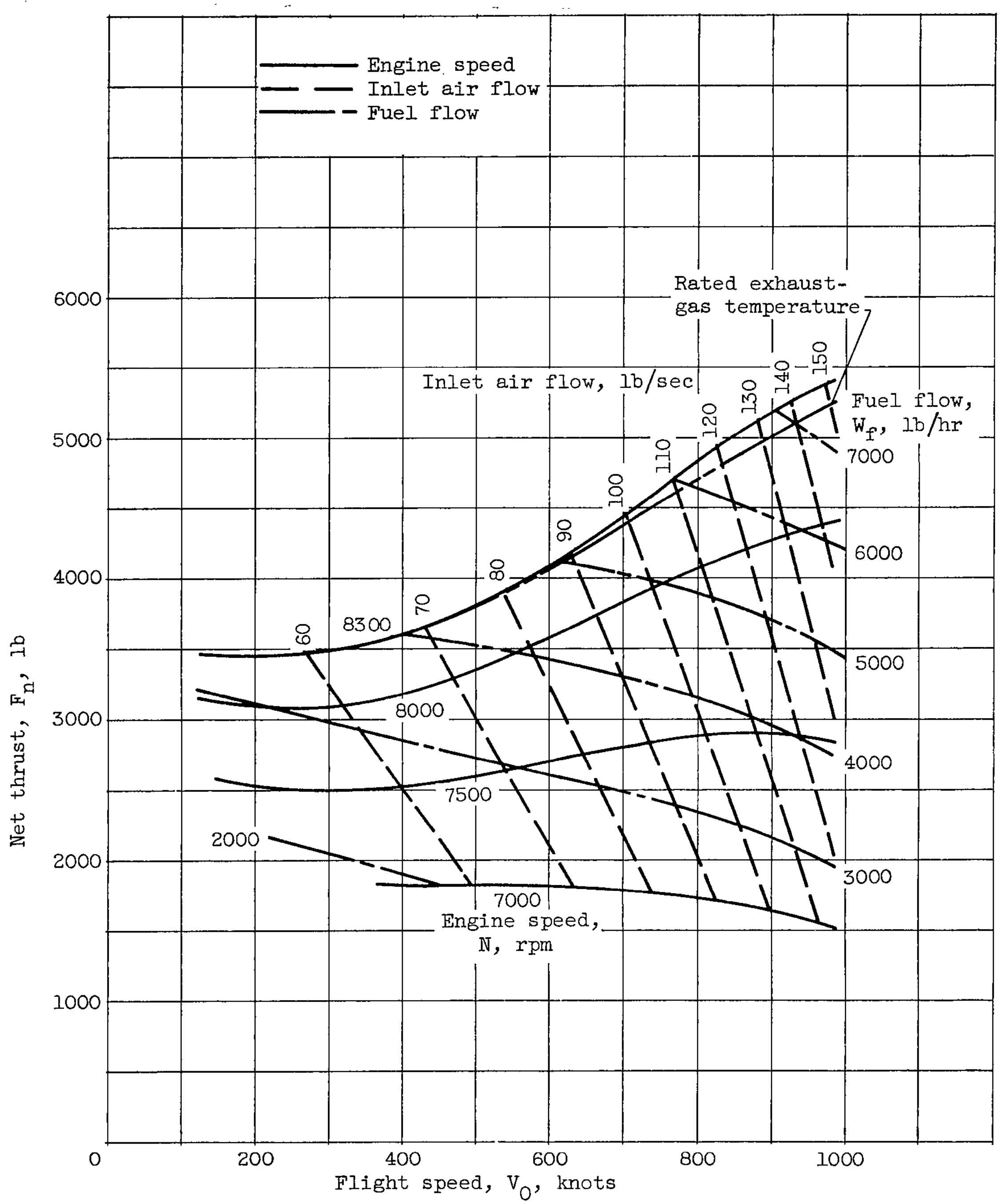
(b) Altitude, 5000 feet.

Figure 11. - Continued. Altitude performance calculated from pumping characteristics for rated exhaust-nozzle area.



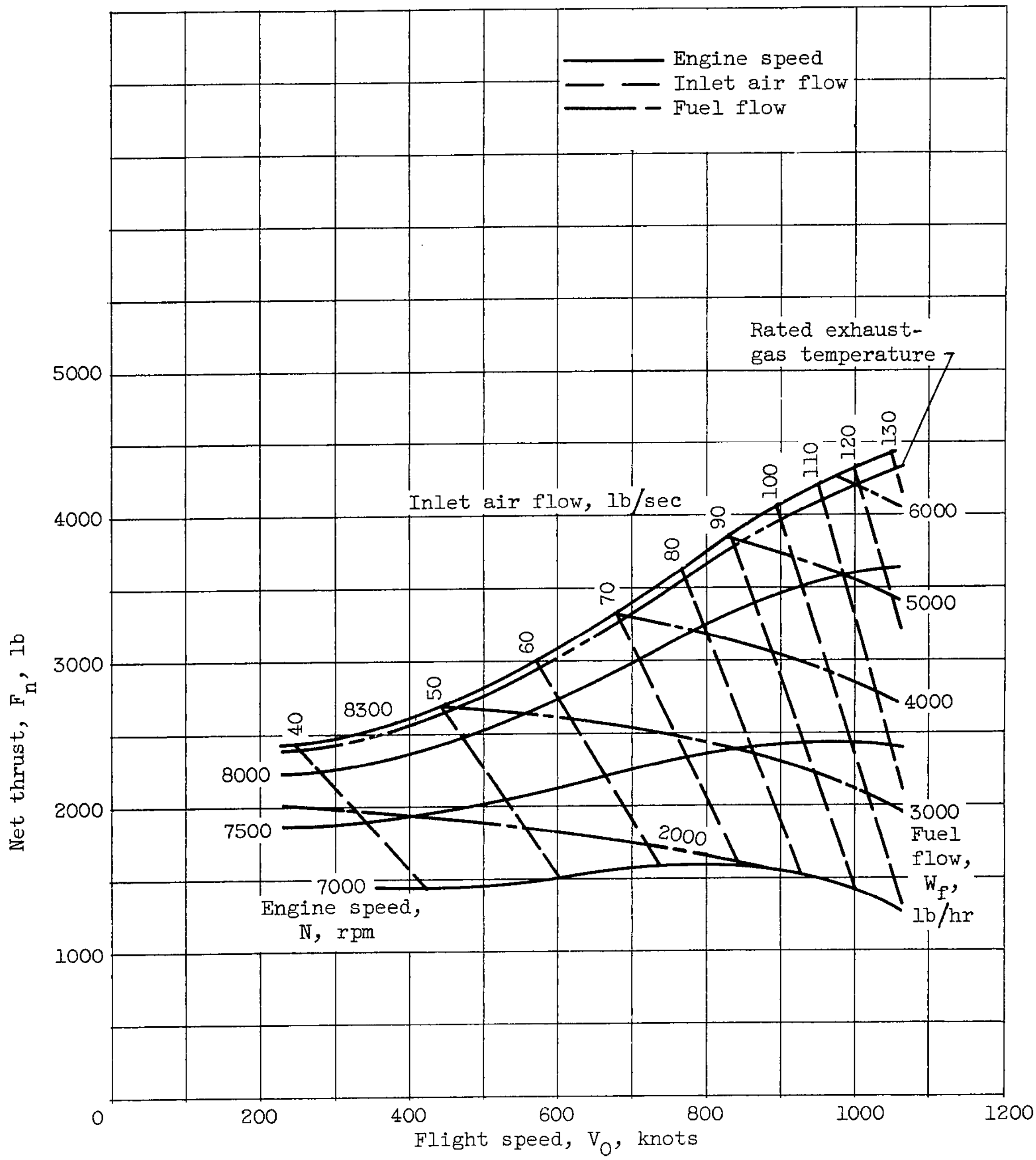
(c) Altitude, 15,000 feet.

Figure 11. - Continued. Altitude performance calculated from pumping characteristics for rated exhaust-nozzle area.



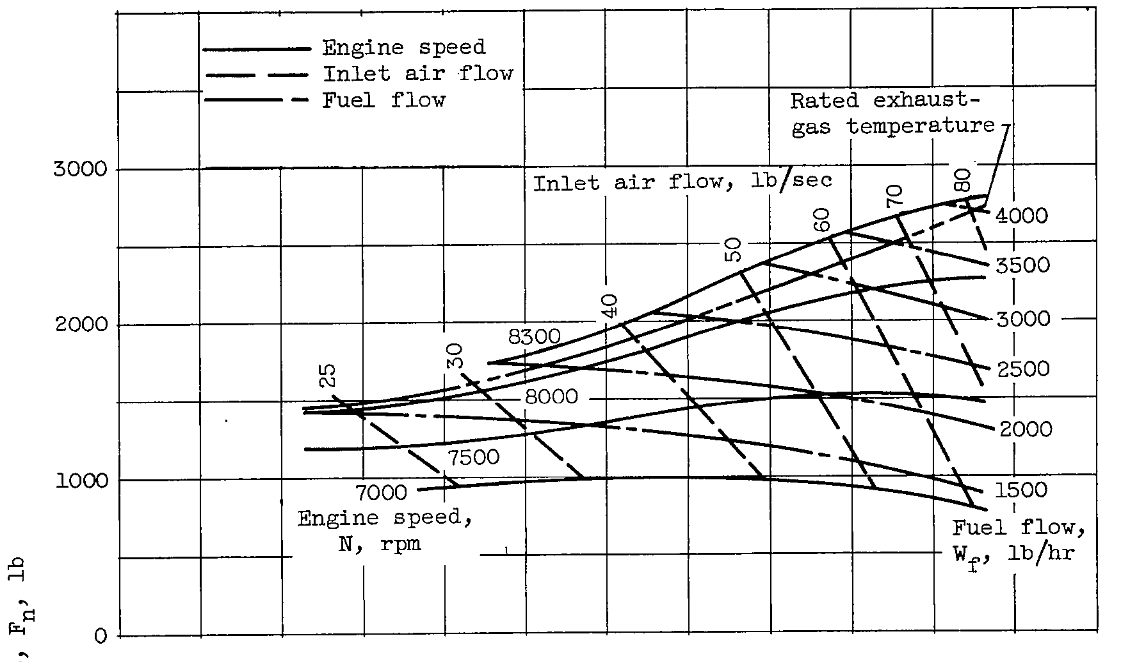
(d) Altitude, 25,000 feet.

Figure 11. - Continued. Altitude performance calculated from pumping characteristics for rated exhaust-nozzle area.

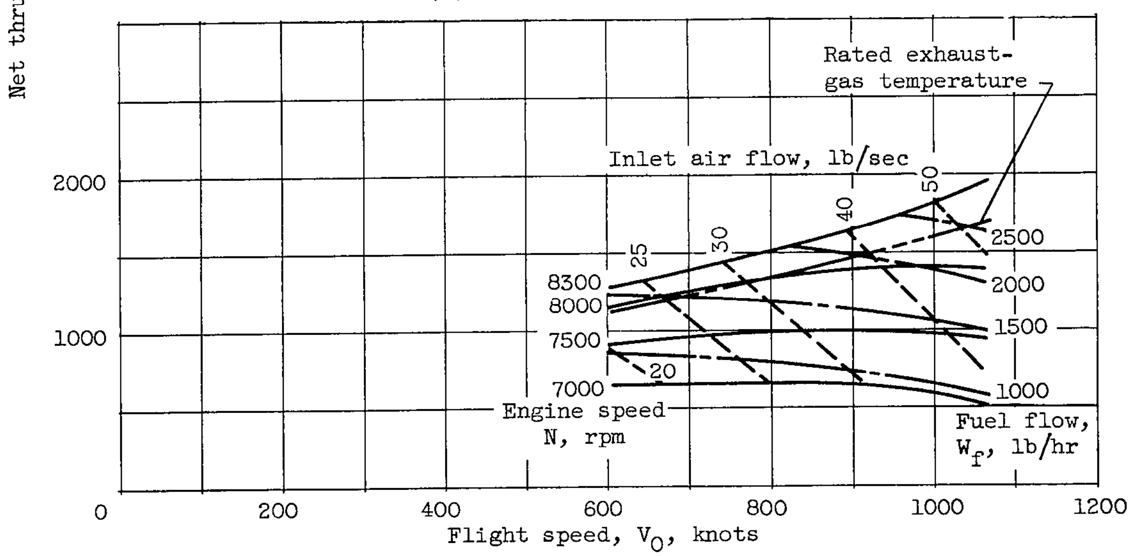


(e) Altitude, 35,000 feet.

Figure 11. - Continued. Altitude performance calculated from pumping characteristics for rated exhaust-nozzle area.



(f) Altitude, 45,000 feet.



(g) Altitude, 55,000 feet.

Figure 11. - Concluded. Altitude performance calculated from pumping characteristics for rated exhaust-nozzle area.

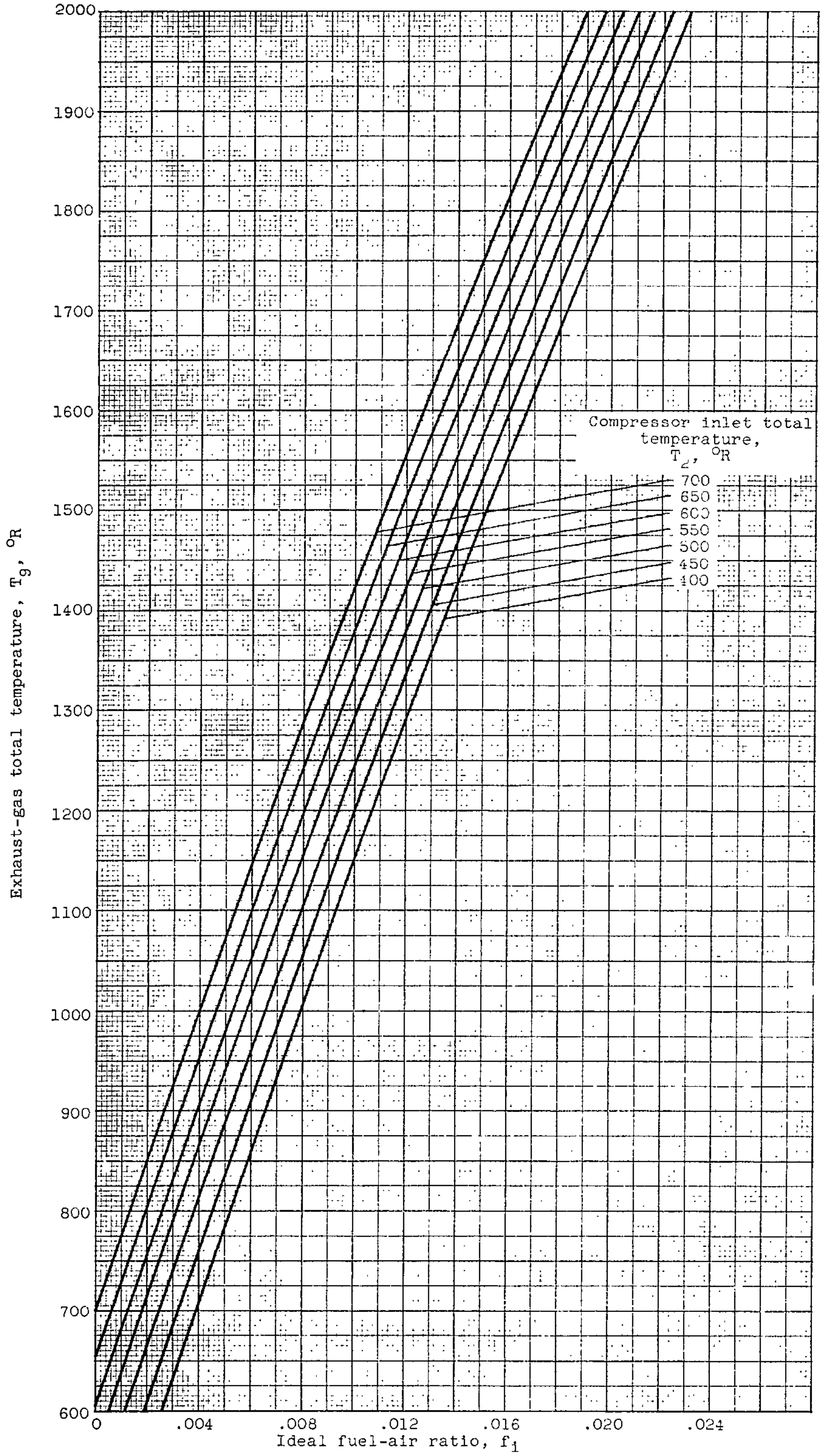


Figure 12. - Ideal fuel-air ratio as a function of engine temperature rise.

OVER-ALL PERFORMANCE OF J65-B3 TURBOJET ENGINE FOR

REYNOLDS NUMBER INDICES FROM 0.8 TO 0.2

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